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U. S. ARMY
TRANSPORTATION RESEARCH COMMAND
FORT EUSTIS, VIRGINIA

TCREC TECHNICAL REPORT 62-73

A STUDY OF THE ORIGIN AND MEANS OF REDUCING HELICOPTER NOISE

Task 9R38-01-022-01

Contract DA 44-177-TC-729

November 1962

prepared by:

BELL HELICOPTER COMPANY
Fort Worth, Texas

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HEADQUARTERS
U. S. ARMY TRANSPORTATION RESEARCH COMMAND
Fort Eustis, Virginia

This report has been reviewed by the U. S. Army Transportation Research Command and is considered to be technically sound. The report is published for the exchange of information and stimulation of ideas.

FOR THE COMMANDER:

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Contract DA 44-177-TC-729
Task 9R 38-01-022-01
November 1962

A STUDY OF THE ORIGIN AND MEANS OF REDUCING HELICOPTER NOISE

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A DIVISION OF BELL AEROSPACE CORPORATION · A TEXTRON COMPANY

For

U. S. ARMY TRANSPORTATION RESEARCH COMMAND
Fort Eustis, Virginia

FOREWORD

This report summarizes the results of an experimental and analytical research program to investigate means of reducing the noise of helicopters. The program was conducted by Bell Helicopter Company under U. S. Army Transportation Research Command Contract DA 44-177-TC-729 (Reference 1) and was carried out under the technical cognizance of Mr. John E. Yeates, USATRECOM, Fort Eustis, Virginia.

The acoustical measurements and the reduction of test data reported herein were conducted by the Acoustical Instrumentation Test Laboratories of General Dynamics Corporation, Fort Worth, Texas. Those data are presented in General Dynamics Report FZM-2471 (Bell Report No. 299-099-192), copies of which are available on request. Personnel associated with this program were Messrs. C. R. Cox, S. M. Hamzeh, J. A. DeTore, J. M. Drees and R. R. Lynn of Bell and Messrs. C. P. Fisher, P. T. Mahaffey, E. L. Kelsey, S. M. White and E. E. Murphy of General Dynamics.

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LIST OF SYMBOLS

a	Velocity of sound, (1117 feet/second, sea level, standard conditions)
b	Number of blades
BL	Blade loading, pounds/square feet
$b\Omega$	Fundamental frequency of rotational noise, radians/second
c	Rotor chord, inches
c_d	Form drag coefficient
C_T	Rotor thrust coefficient, $C_T = \dfrac{T}{\rho \pi \Omega^2 R^4}$
d	Projection of blade profile width on the plane perpendicular to the direction of motion, feet
f	Frequency, cycles per second
GW	Gross weight, pounds
J_{mb}	Bessel function of first kind with index mb
k	Propagation loss coefficient, decibels/1000 feet
K	Strouhal number
M_t	Rotor tip Mach number
m	Order of harmonic
p	Far field sound pressure, pounds/square feet
Q	Torque, foot-pounds
R	Rotor radius, feet
R_e	Effective rotor radius, feet
s_o	Distance from an arbitrary point to noise source, feet
SPL	Sound pressure level, decibels referenced to 0.0002 dynes/square centimeter

T	Rotor thrust, pounds
V	Velocity, feet/second
V_t	Rotor tip speed, feet/second
W_a	Acoustic power, watts
x	Distance from an arbitrary point to the rotor hub along the axis of rotation, positive in the direction of thrust, feet
y	Distance from an arbitrary point to the rotor hub perpendicular to the axis of rotation, feet
α	Angle of attack, degrees
γ	Elevation angle of helicopter from ground observer. degrees
θ	Azimuth angle measured in counterclockwise direction with zero degrees at helicopter tail boom, degrees
μ	Tip speed ratio, $\mu = V\cos\alpha/\Omega R$
ρ	Density of air, slugs/square feet
σ	Rotor solidity
σ	Density ratio
Ω	Angular velocity, radians/second
L_N, phon	Unit of loudness level, referenced to 0.0002 dynes/square centimeter
N, sone	Unit of loudness, $\log_{10} N = 0.03\, L_N - 1.2$ (By definition, a loudness of one sone is selected to correspond to a loudness level of 40 phons) (Reference 9)

I. SUMMARY

This report presents the results of an experimental and analytical study of the origin and possible means of reducing helicopter noise. Acoustical and performance data are presented for a single rotor turbine powered helicopter (HU-1A) with several main rotor configurations. Also, simultaneously recorded rotor blade pressure and acoustical test data are given.

Noise criteria are reviewed and established on the basis of over-all sound pressure and loudness level. The latter pertains to the auditory sensation as perceived by an observer and is preferable as a measure of the relative importance of the various noise sources. Based on the loudness level criterion, the most prominent noise components of the test helicopter are identified. For the far field case, these are, in their order of prominence: (1) main rotor blade slap (when it occurs), (2) tail rotor rotational noise, (3) main rotor vortex and (4) rotational noise and (5) drive system and power plant noise. Blade slap is characterized by high intensity sound pressures of all frequencies and occurs at the blade passage frequency. This noise is shown to be dependent upon flight condition and configuration. It is noted that single rotor helicopters are less susceptible to blade slap than tandem machines.

Main rotor rotational and vortex noise components are defined and identified from the test data. Trends of both rotational and vortex noise are established as a function of the various aerodynamic parameters including thrust, tip speed, number of blades and blade loading. The most significant parameter is shown to be tip speed. The basic trends are noted to be valid for tandem helicopters such as the HC-1B. Similar conclusions are obtained for the tail rotor, based primarily on theoretical considerations. On the basis of these studies, design and operational techniques resulting in minimum far field noise for existing helicopters are given.

Several modifications to the HU-1A and HU-1B helicopters are presented and discussed in relation to their effects on noise, performance and cost. It is shown that significant noise reduction is possible by modifying the main and tail rotors to operate at lower tip speeds.

The best over-all tail rotor modification selected, which includes a new four-bladed rotor with standard HU-1 blades and a new hub and gear box, is shown to have a loudness of about half that of the HU-1A tail rotor. When used on the HU-1A and HU-1B, however, only a 20 per cent reduction in total loudness of the helicopter is realized due to the noise generated by the main rotor. Performance changes resulting from the modified tail rotor are negligible.

The main rotor modifications studied involved increases in blade area to accommodate the low tip speed operation. Both two-bladed and three-bladed high solidity rotors are considered and maximum use is made of existing HU-1A and HU-1B blades. It is shown that the modifications of the main rotor, in conjunction with that of the tail rotor, result in a total loudness reduction for the HU-1A of about 40 per cent. For these modifications, the performance is approximately the same as that of the basic helicopter. Since the tail rotor noise is higher than that of the main rotor, modifications to only the main rotor will result in no appreciable change to the loudness of the noise generated by the helicopter.

It is concluded that for the HU-1 series helicopter, the tail rotor is the noise source which should be first attacked, since it exists during all flight conditions. Blade slap, which is most significant for tandem helicopters, should be further defined and means of its mitigation investigated.

II. CONCLUSIONS AND RECOMMENDATIONS

Based on the results of this study, it is concluded that

1) The most prominent far field noise sources and components for single rotor helicopters of the general size of the HU-1 are, in their order of prominence:

   a) main rotor blade slap (when it occurs).

   b) tail rotor rotational noise.

   c) main rotor vortex noise.

2) Blade slap of a single rotor helicopter is dependent on flight conditions, and consequently its effects may be mitigated by operational techniques. For the tandem helicopter, the problem is more general and can become severe. The mechanism of the generation of this noise is not fully understood; however, indications are that theoretical techniques to describe this source and to evaluate the effects of various parameters are possible. The remaining conclusions are applicable to the general case when blade slap does not occur.

3) The prominent noise sources will vary depending principally on the size of the helicopter; e.g., for small helicopters with high rotor speeds, the main rotor rotational component may be expected to predominate over tail rotor noise.

4) Based on the experimental and analytical data of the subject program, the most significant rotor parameter associated with helicopter noise is tip speed. Lower tip speed operation results in a reduction of all noise associated with the helicopter. The effects of tip speed predominate over the effects of other parameters when they are considered simultaneously.

5) Significant noise reduction of the HU-1A helicopter is indicated to be possible by the use of a new four-bladed tail rotor operating at lower tip speed. Production HU-1 blades and a new tail rotor gear box and hub are required. Based on the results of this study, the loudness of the noise associated with the HU-1A tail rotor may be reduced as much as 50 per cent by this modification. Because of the state of the art of noise prediction, however, this estimate is not certain and should be confirmed by an experimental program.

6) With the new tail rotor, only a 20 per cent reduction in the total loudness associated with the helicopter can be realized due to the noise generated by the main rotor. The reduction in main rotor

noise is dependent on the extent to which the tip speed can be reduced. Reductions are shown for several HU-1 modifications; the most significant involves the use of the HU-1B rotor system on the HU-1A helicopter. For this case, the total loudness of the helicopter is reduced about 40 per cent and the performance is increased slightly. The major effect of the new main rotor is to allow operation at low tip speed.

7) An extension of existing theory relating to prop-rotor noise and additional basic research regarding subjective response to combinations of complex sound pressure waves are needed to define fully and control the noise associated with VTOL aircraft operation.

It is recommended that

1) An experimental verification be made of the noise reduction possibilities indicated herein for the HU-1 tail rotor modification. If the estimated reductions are found to be correct, then consideration should be given to equipping all HU-1 helicopters with that tail rotor.

2) Consideration be given to equipping the HU-1A helicopter with the HU-1B main rotor in addition to the tail rotor resulting from (1) above. For the basic mission investigated, this configuration resulted in the lowest total loudness.

3) An over-all noise control program for VTOL aircraft be initiated to,

   a) develop an adequate theoretical basis for rotor noise prediction, and

   b) verify the theoretical approach by an experimental program to define blade slap and to evaluate the effects on noise of such rotor blade modifications as twist distribution, taper, special outboard shapes, etc.

III. INTRODUCTION

Because of the expanding use of the helicopter for Army tactical missions, for operation in noncombat areas, and for civilian transportation, it has become increasingly important to develop an understanding and means of mitigating the noise associated with VTOL aircraft operation. Recognizing this, the U. S. Army Transportation Research Command initiated this program to investigate means of reducing the noise associated with helicopters and to show the effects of the possible noise reduction modifications on the over-all performance of these machines.

The study was intended to be a general treatment of the helicopter noise problem. It was to be based on the available literature, in addition to the results of a limited acoustical measurement program of several rotor configurations on the HU-1A helicopter. Supplementary measurements of aerodynamic blade pressures were to be used to relate rotor noise with air load variation and blade moments. With these measurements the comparative effects of solidity, thrust coefficient, tip speed, number of blades, etc., were to be obtained directly.

At the onset of the work, a review of the literature confirmed that there are only a few reported investigations (e.g., References 2, 3 and 4) which deal directly with the helicopter noise problem. Further, the theoretical definition of rotor noise is not well developed. Existing theory as used and summarized in Reference 5 relates to the symmetrical flow case associated with a propeller rather than the asymmetrical flow of a rotor, or prop-rotor. In addition, the empirical treatment of high frequency propeller noise of Reference 6 deals only with low Reynolds number flow.

With these limitations in mind, emphasis during the subject program was placed on obtaining acoustical measurements on which ultimately an adequate theory could be based and on developing, insofar as the scope of the program allowed, an understanding of the noise associated with helicopter operation. During the course of the program, established techniques were selected to define quantitatively the principal noise associated with the helicopter and to evaluate the possibilities of its reduction.

The performance, weight, etc., associated with the noise reduction techniques studied are presented for the HU-1 helicopter. Specific modifications to other helicopters are not considered because of the lack of directly comparable data on those machines. For the main rotor, these modifications are based principally on the acoustical data obtained during this program. Propeller theory is used for the tail rotor to establish noise reduction trends associated with various configurations and designs. To evaluate the noise reduction possibilities associated with tail rotor modifications, the calculated data are modified empirically based on the results of the subject program.

IV. EXPERIMENTAL TEST PROGRAM

A. GENERAL

The objectives of the test portion of this program were to provide information on the origin of helicopter noise and to show the effects on noise of the various parameters associated with rotor design. The subject test program included a number of acoustical measurements of the HU-1A helicopter with three main rotor configurations during tiedown, hover and fly-over operation. In addition, aerodynamic pressure measurements on the standard HU-1A configuration during fly-over were made simultaneously with acoustical measurements. A detailed report of the subject test program and the acoustical data are given in Reference 7.

B. DESCRIPTION OF TEST HELICOPTER AND ROTOR CONFIGURATIONS

The basic test vehicle used in this program was the HU-1A helicopter. Several models of the HU-1 airframe were used during the tests of the various rotor configurations. A photograph of the HU-1A is shown by Figure 1. A three-view drawing with dimensional data is presented in Figure 2.

Descriptions of the rotor configurations tested during the program are summarized in Table 1. The various rotors tested are defined in that table as Configurations I, II and III. Configuration I is the standard HU-1A rotor system, Configuration II is the standard HU-1B rotor system, and Configuration III is an experimental three-bladed rotor system. The production HU-1A tail rotor was used with all main rotor configurations.

C. ACOUSTICAL MEASUREMENTS

1. Test Conditions

The test schedule and helicopter operating conditions are given in Table 2. The data of Table 2 include the test and configuration numbers, the rotor configurations and the operating conditions (e.g., engine speed, gross weight, e.g.). Tiedown tests of Configuration I were made at a constant power setting with and without the tail rotor, as noted. Hover and fly-over tests of all configurations are indicated in the table. During the fly-over tests of Configurations I and II, flight conditions of low power letdowns and decelerations were performed and recordings of blade slap were obtained. Changes in thrust and blade loading were achieved by varying the gross weight for each configuration. The range of test gross weights was limited due to weight of the air load instrumentation (Configuration I) and to temporary flight limitations on the experimental three-bladed main rotor (Configuration III).

The environmental conditions encountered during the tests are summarized in Table 3. The wind directions and velocities, temperatures and humidities for the various test dates are presented. The variations in temperature and humidity during the tests were small and only the wind conditions of test runs 21 through 29 (fly-over tests, Configuration II) are considered to be marginal.

2. Microphone Locations

Ground plane measurements were made at radii of 50, 100 and 200 feet and at 30 degrees of azimuth during the tiedown and hover tests. This is illustrated by Figure 3. During the hover tests, the helicopter remained in ground effect (IGE) at approximately 5 feet altitude. Microphone 22 was installed during selected tests inside the helicopter in the pilot's compartment.

Fly-over measurements were made at ground locations with the helicopter flight path at approximately 50 feet altitude. This is illustrated in Figure 4. The microphone locations were on a line perpendicular to the flight path at distances of 100 and 300 feet. The microphones, incorporating wind screens, were attached to aluminum stands with the diaphragm at a height of approximately 5 feet above the ground.

Special Runs (SR) 1 through 6 consisted of varying the microphone height from 5 to 15 feet above the ground plane at distances of 50 and 100 feet in front of the helicopter (Configuration III) to determine the effect of microphone ground height at various distances from the machine. These data are illustrated by Figure 5. It is seen that the over-all sound pressures are greatest near the ground; however, the effect diminishes as distance is increased.

The fly-over measurements were made in an open field with low grass covering. The tiedown tests of Configuration I and the hover tests of Configurations I and II were performed on a concrete ramp. During these tests the microphones were necessarily positioned over concrete and short grass. The hover tests of Configuration III were performed with all microphones located over short grass. The difference in ground covering produced a slig[illegible] variation in measurements.

3. Instrumentation

The acoustical measurements of this program were made with Altec Lansing Equipment including 21BR150 and 21BR180 microphones mounted on Type 165A bases, Type 526A power supplies and Type 420B amplifiers. The output of the amplifier was directed into one channel of an Ampex Type FR-110 Tape Recorder. Figure 6 shows a portion of this equipment. At each microphone location the over-all noise level was monitored and any erratic microphone was replaced. Six microphones were recorded simultaneously with a seventh channel used for identification.

Over-all sound pressure levels were measured and tape recordings of the microphone outputs were made at each position for all operating conditions. A 400-c.p.s. tone of known amplitude and a random noise signal were recorded at the beginning of each series of runs for the purpose of calibration.

4. Data Reduction

The tape recordings taken in the field were played back through the data reduction system shown in Figure 7. The tape recordings were re-recorded in the laboratory on an Ampex FL-100 Loop Recorder. Over-all and one-third octave band levels were recorded on a Bruel and Kjaer Model 2304 Level Recorder. Narrow band analyses were made with a 6-c.p.s. constant bandwidth filter, the output being recorded on a Moseley Autograph X-Y Recorder (frequency versus sound pressure level).

The data reduction schedule for the tests is shown in Table 4. Over-all levels were recorded for all test conditions. One-third octave analyses of one hover condition were performed for each configuration, and 6-c.p.s. bandwidth analyses of the recorded data were made for a number of microphone locations. In addition, oscillograph time histories of selected data were performed using a 25-c.p.s. constant bandwidth filter.

The readings taken from the tape were corrected for filter insertion loss and were adjusted for recording and playback attenuation settings and system gain factors. Corrections for the frequency response of the two types of microphones used during the measurements are given in Figure 8. These corrections are to be applied to all 6-c.p.s. constant bandwidth data presented in this report and in Reference 7. In addition, these data in the 10-to 50-c.p.s. frequency range were found to be shifted slightly from the correct frequency. This results in the low frequency main rotor noise components being displaced several c.p.s. from their actual [illegible]. At approximately 70 c.p.s. an erratic system resonance was found to occur. This was due to the data reduction system and is not present in the noise spectrum; therefore, this peak was disregarded.

Data reduction by 6-c.p.s. constant bandwidth analyses offers the advantage of accurate resolution of noise composed of discrete frequency components. However, attention must be given to the dynamic range of the acoustical instrumentation system and to the noise rejection properties of the filter when analyzing noise with high intensity, low frequency components (typical of helicopter noise). For the majority of the subject tests, the upper and lower limits of the dynamic range were approximately 105 and 65 decibels. Therefore, noise components with levels below this lower limit are not identified. In addition, the maximum rejection of the 6-c.p.s. filter to noise outside the bandpass is approximately 25 decibels, therefore, high frequency components with levels of 25 or more decibels below that of the low frequency main rotor noise are not resolved.

D. AIR LOAD MEASUREMENTS

Differential pressures on a rotor blade were measured simultaneously with acoustical data to correlate the noise generated by a helicopter main rotor with the aerodynamic loads on the blade. Both air load data and internal and external noise measurements were recorded for the fly-over test conditions of Configuration I (test runs 30 through 42).

The measurement of main rotor aerodynamic loads was accomplished by the use of NACA-type pressure transducers mounted inside the blade. These transducers were mounted at five radial distances, 40, 75, 85, 90 and 95 per cent radius, and seven chord locations extending from 2 per cent to 90 per cent chord. Before and after each test, each transducer was calibrated with a reference differential pressure.

The air load data measured during the subject program for two operating conditions are presented in the Appendix. Portions of these data are used in later sections to relate main rotor generated noise with differential pressure fluctuations over the blade.

It was intended to correlate blade bending moments, pressures and rotor noise simultaneously; however, the test helicopter (from another program) was not in the proper configuration for measuring blade bending moments at the time the subject tests were conducted. For the interested reader, similar flight conditions with blade bending moments and differential pressures are reported in Reference 8.

V. DISCUSSION OF HELICOPTER NOISE AND RESULTS OF TEST PROGRAM

A. GENERAL

The noise produced by the operation of a helicopter is an unavoidable consequence of propulsion. Because of the multiplicity of components of a helicopter propulsion system, an observer detects a complex combination of sound energies. This combination results in the noise associated with helicopters which unfortunately cannot be eliminated completely. It is believed, however, that future helicopters may be designed to produce acceptable noise levels for most missions.

In initiating a noise control program, an understanding of the origin of the noise related to the various sources is required. Such an understanding must include the identification of the various sources, the description of the generation of that noise, and an evaluation by a meaningful criterion of the sounds which an observer perceives as the prominent noise produced by the vehicle. The following paragraphs discuss these items for the helicopter and are aimed at presenting a general treatment of the problem based on the results of the subject test program and the findings of this study.

B. CRITERIA

Noise is defined as an undesirable or unwanted sound. Sound is composed of pressure waves whose magnitudes and frequencies are sensed by the human ear. The undesirability associated with the sound involves the subjective response of the observer which includes not only the physical stimulus of the ear as a function of intensity and frequency of the perceived noise, but also psychological factors. Thus, the observer perceives the noise in terms of whether or not the sound is loud, annoying, interferring with his speech, etc. In effect, the observer instinctively establishes a criterion by which he judges the acceptability of a noise.

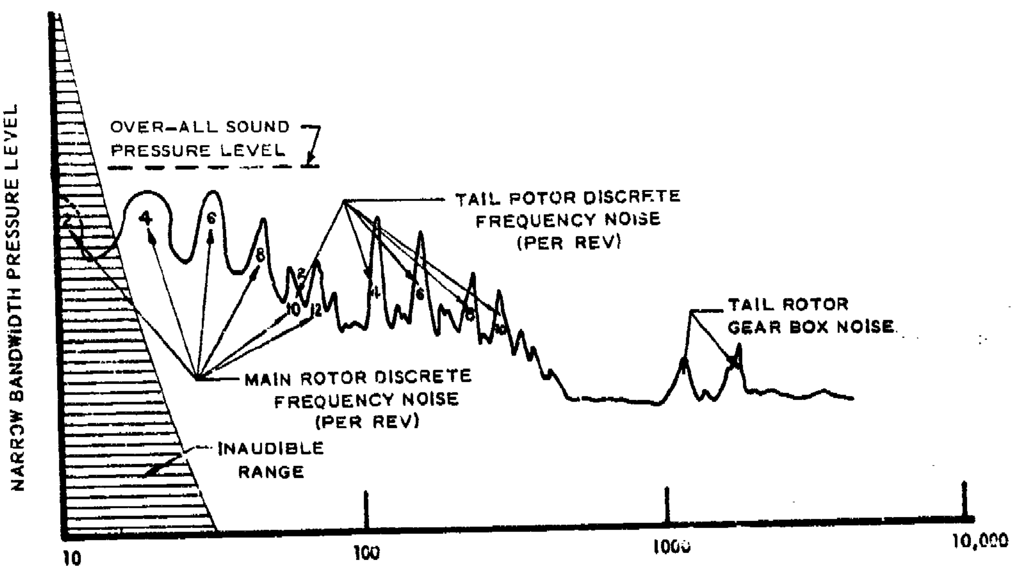
The criteria used during the subject program to evaluate the possible reduction of helicopter noise are based on over-all and component sound pressure levels and component loudness levels. A brief discussion of each is given below in addition to comments on the character of helicopter noise.

1. Over-all Sound Pressure Level

Over-all sound pressure levels consist of the total sound pressure intensity reaching an observer; however, this measure does not include the effects of frequency. These levels are used to obtain [illegible] indication of the over-all magnitude of a given noise; however, [illegible] in evaluating quantitatively or comparing noise of different type and/or character. Over-all sound pressure levels are measured in decibels and are logarithims of the sound pressure referenced to 0.0002 dynes/square centimeter. Unless otherwise noted, all acoustical data presented in this report are referenced to this sound pressure.

2. Component Sound Pressure Level

Additional information is obtained from a narrow bandwidth analysis that shows the sound pressure level as a function of frequency. From this, the sound pressure level emanating from the major sources may be identified. The accompanying sketch illustrates this. Note that the noise of the main and tail rotors and transmissions may be identified. It is also indicated on the sketch that the over-all sound pressure level is almost exclusively determined by the main rotor two-, four- and six-per-rev noise peaks.



NARROW BANDWIDTH ANALYSIS
OF HELICOPTER NOISE

Although the investigation of the frequency spectrum permits a determination of the components contributing to the noise, it is inadequate to define the noise as perceived by an observer. Below about 20 c.p.s. and above about 20,000 c.p.s., sound is inaudible. Because the highest sound pressures associated with helicopters occur at low frequencies (11, 22, etc. c.p.s.), a criterion based only on sound pressure levels will result in an incorrect evaluation of the relative prominence of the noise produced by the various components. To alleviate this, a criterion based on component loudness level is used.

3. Component Loudness Level

The loudness of a noise pertains to the magnitude of the auditory sensation experienced by the observer. A measure of the loudness of the various component sources may be obtained from loudness levels determined from narrow bandwidth sound pressure levels and the empirical data of Reference 9. In that reference, loudness levels expressed in phons of single frequency tones covering the audible range of frequencies are defined as curves of equal loudness level on a sound pressure-frequency plot. Loudness level contours from Reference 9 are shown by Figure 9. These curves show the free field sound pressure level of tones of different frequencies judged to be equally loud. The unit of loudness level, the phon, is referenced to 0.0002 dynes/square centimeter similar to sound pressure level.

It should be noted that both sound pressure level and loudness level are logarithmic functions and the comparative loudness levels of various sources are not additive. To add and compare noise on a linear scale, another unit is introduced in the literature and is termed the "sone." Figure 10 is a nomogram for converting loudness levels to loudness, in sones. Direct comparison of loudness may be made on the sone rather than the phon scale for noise of widely separated frequency components.

4. Character

In evaluating helicopter rotor noise, an important consideration not included in the above criteria is the character of the noise. The character of a particular noise source influences the detectability of that source; thus, a pulsating or modulating sound will be easier to detect than a steady noise of the same loudness level. Such is the case for main rotor noise which periodically increases and decreases in intensity due to rotating sources. The character of a noise is not directly considered in the loudness level criterion; however, an attempt to approximate this effect is made by evaluating the peak levels for the rotor noise components instead of the average or root-mean-square values.

C. OVER-ALL NOISE

The over-all noise of a helicopter varies for different modes of operation. During hover, the over-all sound pressure at the center of the rotor is constant except for the effects of small control motions and downwash disturbances. During forward flight, the ground observer detects a variation in the over-all noise due to the asymmetrical air loads acting on the rotor, to distance and Doppler effects.

To establish the over-all noise characteristics of the test helicopter under controlled conditions, over-all sound pressure level measurements were taken while the machine was operated on tiedown. These tests were conducted to establish the relative intensities of the helicopter noise at different angular and radial positions with respect to the machine.

Hovering acoustical tests were also conducted for the purpose of comparing all rotor configurations while operating in a normal flight mode. Additionally, fly-over tests were conducted to evaluate motion and changing distance effects and to study the in-flight characteristics of helicopter noise.

The following paragraphs present the results of the tiedown, hover and fly-over over-all sound pressure level measurements of the subject program. Also, the possibilities of the reduction of the over-all sound pressure of the test helicopters are discussed.

1. Tiedown

The distribution of the over-all noise around the HU-1A (Configuration I) is shown by Figure 11. Near the machine, the over-all sound pressure level in the aft left hand quadrant is the highest. Farther removed from the machine, the over-all sound pressure levels at a given distance are essentially constant.

The effect of disconnecting the tail rotor is shown by a similar plot in Figure 12. A reduction in the aft left hand quadrant sound pressure levels results, and it is seen that the main rotor dominates the over-all far field noise. These data show that the directivity pattern of the tail rotor influences the near field over-all noise of the helicopter.

The over-all sound pressure levels at all microphone locations for the tiedown and hover tests are shown in Table 5. It can be seen that the rotor (engine) speed is of prime importance in establishing the over-all sound pressure level. Decreasing the engine speed from 6700 to 5800 r.p.m., results in a maximum sound pressure level reduction of 9 decibels from 93 decibels (compare tests 5 and 7).

2. Hover

The over-all far field sound pressure levels of the rotor configurations tested are shown by Figure 13 as a function of blade loading and tip speed. For equal thrust conditions, test points taken from Table 2 are shown. Through these points, curves of constant tip speed are faired. For reference, lines of constant thrust coefficient/solidity (proportional to mean blade lift coefficient) are superimposed on the plot.

These data show that for the range of parameters tested, the over-all sound pressure level will approach a minimum at blade loadings near 60 pounds per square foot. Above that value, the noise level will increase with increased blade loading. Doubling the blade loading from 60 to 120 pounds per square foot will increase the noise level approximately the same as will a 10 per cent increase in tip speed (3 decibel increase from 89 decibels). For the conditions tested, it can be seen that tip speed has a significant effect on the over-all noise. The effect of number of blades cannot be determined from these data.

The data indicate that it is improbable that the over-all external noise level of the HU-1A at 200 feet can be reduced to 75 decibels (Reference 1) by a practical change in rotor parameters. This is due mainly to the contribution of the high intensity, low frequency rotor noise.

It should be noted that these data were obtained for conditions during which no significant stall or compressibility phenomena were encountered. It is believed that the general form of the data will remain valid for different operating conditions; however, the magnitude of the sound pressure level would be expected to increase with increased thrust, stall or compressibility.

3. Fly-Over

Sample time histories of the fly-over noise produced by the various rotor configurations are shown by Figure 14. These traces show two major frequency components which correspond to the blade passage frequencies (two-per-rev) of the main and tail rotor. It may also be noted that as the helicopter flies over, a rapid decrease in the over-all sound pressure level results, and the lower frequency main rotor component becomes masked by the tail rotor noise. The increase in the tail rotor noise as the helicopter flies by is characteristic of single rotor helicopters and is a result of the varying sound pressure associated with the moving directivity pattern of both the main and tail rotor noise components, plus Doppler effects.

Table 6 lists the conditions and results of the fly-over tests. The effects of distance on the fly-over noise of the three test configurations are shown, and it can be seen that the three-bladed main rotor configuration produced the lowest over-all sound pressure level (103 decibels at fly-over). This is in agreement with the findings of the hover tests. For higher forward speeds of 90 to 105 knots and an increase in tip speed of 753 f.p.s., the levels shown in Table 6 increase two to three decibels.

It is apparent that only limited information regarding the effects of the various possible aerodynamic parameters and other noise reduction techniques can be obtained by using over-all sound pressure levels as a criterion. As stated previously, the over-all sound pressure level is determined almost solely by the low frequency noise components of the main rotor and, therefore, reveals nothing of the contributions of higher frequency noise sources. To assure a more meaningful criterion, component loudness levels as discussed earlier are used during the remainder of this study.

D. ORIGIN AND CONTROL OF HELICOPTER NOISE

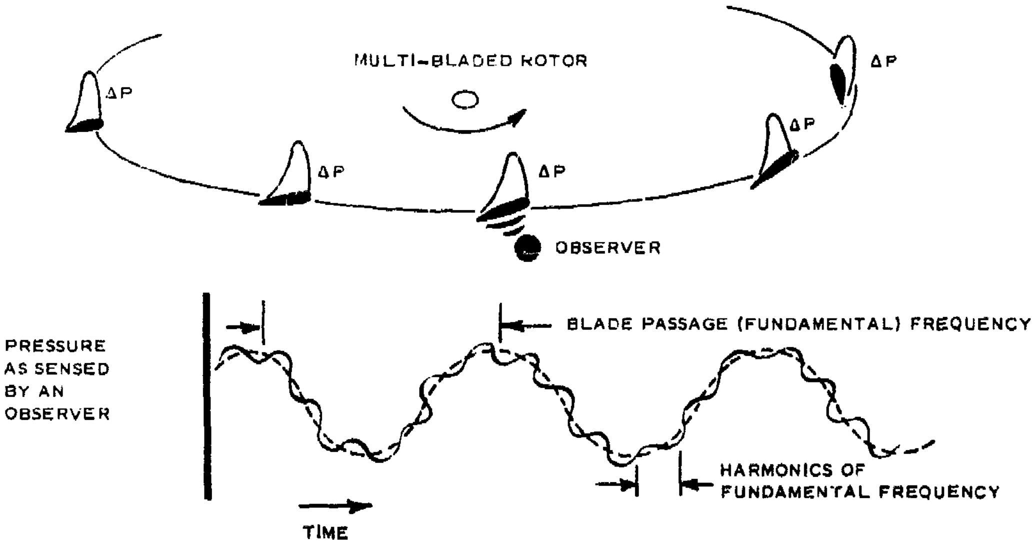
The aggregate of all component noise emanating from a helicopter gives that machine its characteristic acoustical signature. That signature is not only a function of the specific noise sources of the vehicle but also depends upon factors such as distance, terrain, ground cover, etc., as well as the response of the observer.

In this section, the noise sources associated with helicopter operation are identified and explained insofar as possible, and the various factors which influence the acoustical signature of the machine are given. The noise sources are presented and discussed in relation to that part of the vehicle's propulsion system from which they emanate (rotors, drive system, power plant, etc.). The effects of the various parameters associated with each source are discussed and trends of possible noise reduction are established. This section is generally applicable to both tandem and single rotor helicopters.

1. Rotor System(s)

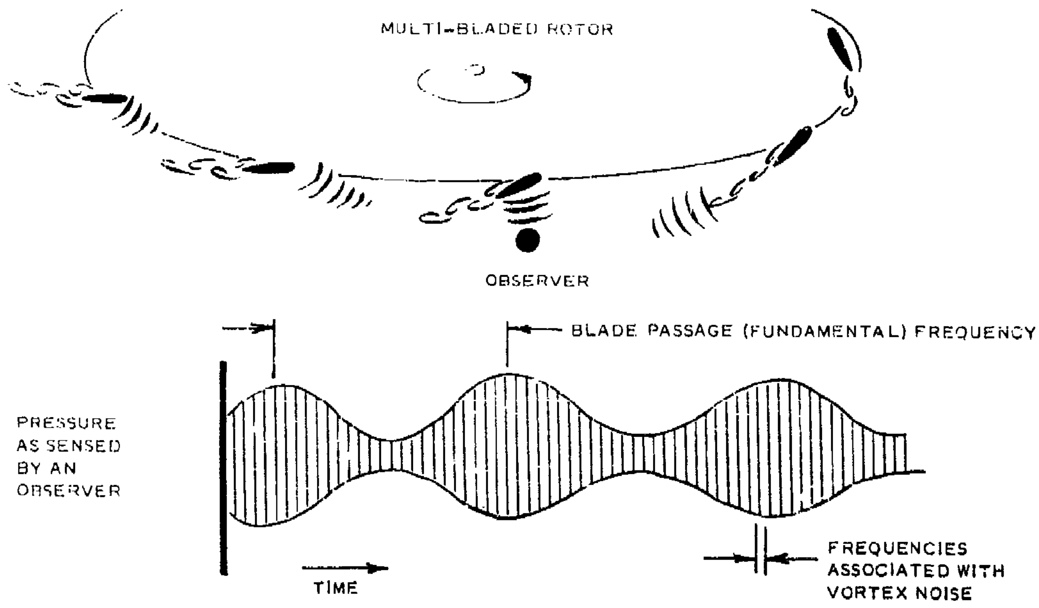
Rotor noise is produced by both aerodynamic forces and structural vibrations. Except in unusual cases (such as stall flutter), the noise of aerodynamic origin is by far the most important. Consequently, this section will only be concerned with rotor aerodynamic noise.

In the literature, propeller (rotor) noise is divided into two components, rotational and vortex noise. Rotational noise is composed of discrete frequencies which are multiples of the blade passage frequency and is associated with the total thrust and torque of the blades. As represented by the sketch below, an observer off the axis of rotation senses a variation in pressure due to the rotation of the blades and their surface pressure distribution (ΔP). This variation is associated with the blade passage or fundamental frequency and its harmonics. If the blade passage frequency is sufficiently low, several of the low harmonic pressure pulses are not audible.



ROTATIONAL COMPONENT OF PROP-ROTOR NOISE

Rotor or prop vortex noise results from the stresses acting on the medium, i.e., the boundary layer shear and the stresses arising from the wake vortices, and extends over a large range of frequencies defined by the local air flow and the frontal area of the blade. A simplified representation of this component is shown by the following sketch. The distinguishing characteristic of vortex noise is the amplitude modulation of sound pressure at the blade passage frequency.



VORTEX COMPONENT OF PROP-ROTOR NOISE

The actual description of prop-rotor noise is far more complicated than represented here. This is due to Doppler effects, directivity patterns, attenuation factors, etc.

A noise produced by a rotor which is not treated in the literature is the sometimes severe blade slap. Depending on the intensity of the noise, it is described as a "popping" or "cracking" sound. The severity of this noise depends on the rotor parameters and configuration; however, even for the same helicopter, it will vary in intensity depending on flight conditions. Despite its elusiveness, when severe blade slap or crack occurs, it is the predominant noise associated with helicopter operation. A discussion of the above mechanisms and their significance for the main and tail rotors of helicopters is given below.

a. <u>Theoretical Development</u> - To date, the theoretical considerations relating to rotor noise are limited to idealized uniform force distributions over the disc and to diameters associated with propeller design. The following discussions are presented to illustrate the present state of the art of rotor noise prediction.

(1) Rotational Noise - A rotating blade with its associated lift and drag force distribution exerts equal and opposite reaction forces on the air. These forces cause elastic depressions of the air which are transmitted as pressure waves at frequencies determined by the air load or force variation. At a fixed point in space, the fundamental frequency of these pressure waves corresponds to the blade passage frequency.

The development of the theory of propeller rotational noise is given by References 5, 10 and 11. From Reference 5, the far field sound pressure, p, for a propeller at zero forward speed is given by the expression:

$$\left| p \right| = \frac{mb\,\Omega}{2\,\pi\,a s_o} \quad \frac{1}{R_e} \left| Tx \; \frac{R_e}{s_o} \; - \; Q \, \frac{a}{\Omega \; R_e} \right| \quad J_{mb} \left(\frac{mb\,\Omega\;y\;R_e}{a s_o} \right) \quad (1)$$

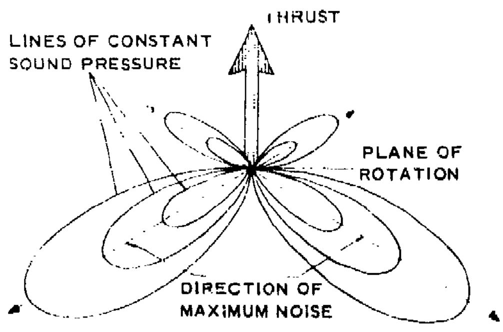
where m is the order of harmonic, b is the number of blades, Ω is the rotor speed, a is the velocity of sound, s_o is the distance from the noise source to the observer, T is the thrust, x is the distance from the observer to the noise source along the axis of rotation, R_e is the effective radius, Q is the torque, J_{mb} is the Bessel function of first kind with index mb, and y is the distance from the observer to the axis of rotation.

By careful examination of Equation (1) it can be seen that the rotational noise is primarily a function of the total thrust produced by the propeller. As the number of blades is increased, the Bessel function tends to decrease and results in a lower noise intensity.

It can be noted that the torque term $Q \frac{a}{\Omega R_e}$ is independent of the position of the observer, whereas the thrust term $Tx \frac{R_e}{s_o}$ is dependent on the observer's location relative to the propeller plane of rotation.

The thrust term is positive for distances in front (i.e., in the direction of thrust) and negative for distances aft of the plane of rotation. Thus, a varying sound pressure will be observed at various distances along the axis of rotation and a rotational noise directivity pattern is defined.

The sketch below illustrates the directivity pattern of rotational noise for a two-bladed propeller in the axial flow condition. Lines of constant sound pressure are shown in relation to the plane of rotation. The maximum noise is seen to be in the direction of the inflow. This pattern rotates with the blades.



DIRECTIVITY PATTERN OF PROP–ROTOR ROTATIONAL NOISE

The principal limitations of the propeller theory involve the assumption of a rectangular chordwise pressure distribution and the resolution of the spanwise distribution of thrust into a constant value acting at one radial station. To eliminate these restrictive assumptions for the helicopter rotor case, extension of the theory is required to include the effects of variation in the air loads as a function of blade azimuth position, radial location, forward speed, Mach number, stall, etc. The work of References 5, 8 and 12 should prove helpful in forming the basis for this extension.

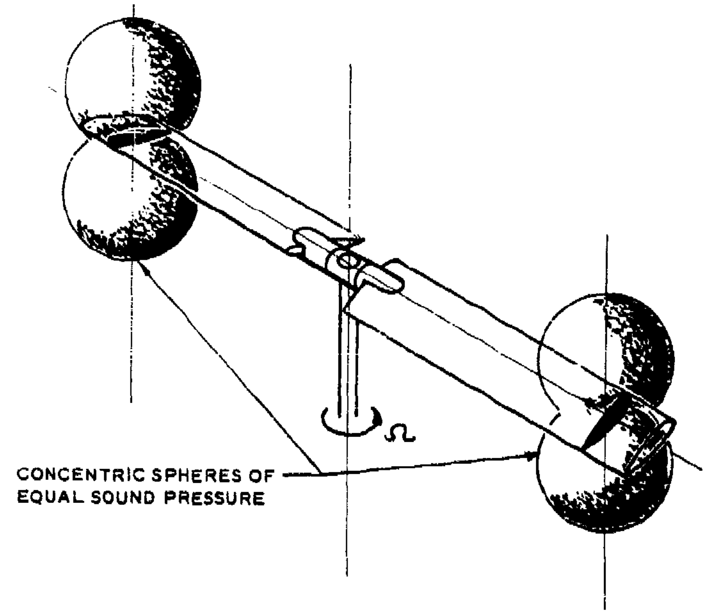
(2) Vortex Noise – Another type of sound radiated from a propeller is termed vortex noise, defined as that due to the shedding of vorticity. This noise is caused by the stresses (Reynolds, hydrostatic, viscous, etc.) acting on the medium and its treatment in the literature has been primarily empirical. Reference 6 gives a proportionality relationship for the vortex acoustic power, W_a, radiated by a rotating cylindrical rod as:

$$W_a \propto \frac{\rho}{a^3} (c_d{}^2 K)^2 V_t{}^6 Rd \qquad (2)$$

where c_d is the form drag coefficient at the mean radius, K is Strouhal's number, V_t is the tip speed, R is the blade radius, and d is the frontal profile width. With c_d and K independent of velocity, the acoustic power is proportional to V_t^6; since the sound pressure, p, is proportional to $W_a^{1/2}$, then $p \propto V_t^3$. Thus, at low tip Mach numbers and low Reynolds numbers, the blade tip speed is the most influential parameter affecting vortex noise. It is to be emphasized that Equation (4) is at best an approximate relationship.

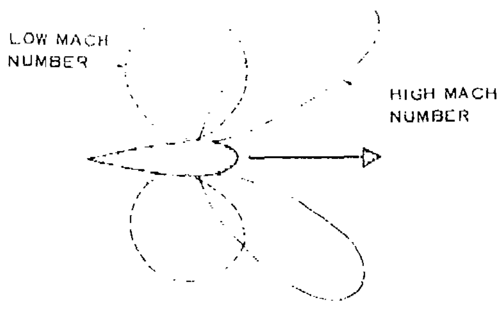
The predominant frequency of vortex noise has been defined empirically as $f = K \frac{V}{d}$ where V is the elemental flow velocity. Consequently, a range of frequencies proportional to the flow velocities over the blade surfaces will be generated by all of the elements of the blade.

At low blade tip speeds, the vortex noise component of each blade element has a directivity pattern as indicated in the sketch below. At each blade element, sound eminates as concentric



DIRECTIVITY PATTERN OF PROP-ROTOR VORTEX NOISE

spheres as a function of the local stresses on the medium. For a small diameter propeller, these spheres can be considered concentrated at the three-quarter radial position. For large diameter rotors, however, no simple representation has been formulated due to interference effects, intensity variation, etc.



EFFECT OF MACH NUMBER ON VORTEX NOISE DIRECTIVITY PATTERN

At high tip speeds, Mach number effects apparently cause a distortion of the vortex noise directivity pattern so as to elongate the spheres and move their upper end toward the direction of rotation. (See adjacent sketch). This effect is treated extensively in Reference 13. Since the vortex noise and its directivity pattern rotate with the blade, an observer off the axis of rotation senses a varying sound pressure. Thus, the observer perceives a modulation of the vortex noise at the blade passage frequency.

The situation in the case of a helicopter rotor is much more complicated than for a propeller. Boundary layer and induced vortices are continuously being shed from the blades due to factors such as the angle of attack variation and blade motions. To predict vortex noise with any degree of accuracy, detailed information on the high frequency aerodynamic loads on the blade and fluctuating stresses in the boundary layer are required. Theoretical refinements of rotor aerodynamics such as those offered by References 14 and 15 may be of considerable importance in this respect.

b. Main Rotor

(1) Rotational and Vortex Noise

(a) Identification of Main Rotor Noise - Figure 15 shows the far field noise spectrum of the HU-1A test helicopter as defined by constant bandwidth analysis. It can be seen that on the basis of component sound pressure level the main rotor rotational noise is predominant. At the higher frequencies, the principal noise components are the tail rotor rotational and the main rotor vortex noise. Other identifiable sources are those associated with the tail rotor gear boxes.

Rotational noise components are identified by their characteristic frequencies (multiples of the blade passage frequency). Although for convenience, the various harmonics of the rotational noise are represented in later sections of this report by an envelope, rotational noise contains only discrete frequencies. The identification of vortex noise requires a more detailed analysis of the constant bandwidth data.

In an earlier section, it was pointed out that rotor vortex noise occurs at frequencies up to a maximum defined by the blade tip speed. Consequently, vortex noise may be properly represented by an envelope. Since the directivity pattern of this noise rotates with the blade, an observer off the axis of rotation will detect the vortex noise at frequencies which are modulated at the blade passage frequency.

Detailed analyses of the constant bandwidth data were made in the frequency range where main rotor vortex noise components were anticipated. Figure 16 shows typical oscillograph traces of vortex noise time histories obtained using a 25-c.p.s. constant bandwidth filter centered at 100, 200 and 300 c.p.s. It is seen that the two-per-rev (11-c.p.s.) modulation occurs at all three frequencies and is clearly evident at 200 c.p.s. Similar analyses at the principal tail rotor rotational frequencies did not show the two-per-rev modulation, thus the levels of this component mask the main rotor vortex noise.

With the various components so identified, Figure 17 presents the envelope of the main and tail rotor noise components and also curves of equal loudness level. It is seen that even though the sound pressure levels of the main rotor rotational noise are highest, the corresponding loudness levels are lower because of their nearly inaudible, low frequency components.

Figure 18 illustrates this effect more directly. On that figure, the loudness levels of the main and tail rotor components are shown as a function of frequency. It is immediately seen that with respect to the noise perceived by an observer, the tail rotor rotational and main rotor vortex noise are the loudest.

(b) Effects of Rotor Parameters - Main rotor loudness levels for the various rotor configurations tested are shown by Figure 19. For these data, the major parameters are constant except blade loading and the number of blades. It should be noted that the gross weight of Configuration III was approximately 5 per cent lower than that for either Configuration I or II. Also, Configuration I had 12 degrees blade twist and an NACA 0015 airfoil section, while Configurations II and III had NACA 0012 sections and approximately 10 degrees blade twist. It can be seen that the main rotor vortex noise predominates for all configurations tested.

Figures 20 and 21 illustrate the effects of blade loading and/or thrust at constant tip speed for Configurations I and II, respectively. In both cases, increasing the thrust resulted in increased rotational and vortex noise. (Although changes in gross weight are used throughout this report to indicate the effect of thrust, the total thrust equals the gross weight plus the fuselage download.)

Figures 22 and 23 show the effects of tip speed for the two- and three-bladed rotors, respectively. The significant points to note are that the loudness level of all noise components increase with increased tip speed and the three-bladed rotor produces the lowest vortex noise of the three configurations tested.

The peak vortex and rotational loudness levels from Figures 19 through 23 are used in the following paragraphs to establish trends associated with the parameter variations of the subject tests. It is noted that the application of these data beyond the range of this investigation, or broad generalizations based on such presentations, may not be valid due to such items as helicopter size effects on the fundamental frequency; the variations of the intensity of the higher harmonic rotational noise; and changes in the characteristics of the modulated vortex noise.

1) Thrust, Blade Loading and Number of Blades - Figure 24 shows the peak values of the main rotor rotational and vortex loudness levels plotted as a function of blade loading for several gross weights. The curves are based on the two-bladed rotor data of Figures 20 and 21 and on the three-bladed rotor data of Figure 23.

The effect of thrust is to increase the loudness level of both the rotational and vortex components. As shown in Figure 24a, the change in the vortex loudness level for both the two- and three-bladed rotors due to a change in thrust remains essentially constant as a function of blade loading. For the rotational component, however, it appears that the effect of thrust for two-bladed rotors is more significant than that for three-bladed rotors at the lower blade loadings.

Increasing the blade loading increases the vortex component loudness level and decreases that of the rotational component. Comparing Figures 24a and 24b, it can be seen that above blade loadings of approximately 50 and 70 pounds per square foot, the vortex component predominates for the three- and two-bladed rotors, respectively.

The three-bladed rotor data shown in Figure 24 are interesting since they allow some observations on the effects of the number of blades. For a given thrust, the loudness levels of the vortex component for the three-bladed rotor fall in line with the extrapolated two-bladed rotor data. This indicates that for conditions of equal hovering efficiency, the vortex noise of two- and three-bladed rotors is essentially the same. In Figure 24B, the peak rotational loudness level of the three-bladed main rotor is appreciably lower than that for the extrapolated two-bladed rotor data (for the same rotor thrust) and consequently indicates a significant effect of number of blades. For the case shown, increasing the number of blades from two to three appears to be equivalent to decreasing the thrust of the two-bladed rotor by about 500 pounds.

In many cases, two-bladed rotors are designed with higher blade loadings than comparable three-bladed rotors. Consequently, the rotational loudness levels of the two- and three-bladed rotors will be approximately the same. Vortex noise, on the other hand, will generally be more prominent for the two-bladed rotor designs.

2) Tip Speed - Figure 25 shows the peak values of the main rotor rotational and vortex loudness levels plotted as a function of blade tip speed. Both two- and three-bladed data are given; the three-bladed data are for hover at a gross weight of 6400 pounds (BL = 50 pounds per square feet) and the two-bladed data are for tiedown at an approximate thrust of 6000 pounds (BL = 108 pounds per square feet).

It is seen that both the rotational and vortex loudness levels increase with increased tip speed and that the respective slopes of loudness level versus tip speed for both components for the two- and three-bladed rotors are the same. The slope of the rotational component is slightly greater than that of the vortex noise, as would be expected from theoretical considerations. The relative magnitudes of the rotational and vortex loudness levels for the two- and three-bladed rotors are explained by the difference in blade loading between the test rotors (see Figure 24).

(c) Reduction of Main Rotor Noise - From the above, it is seen that for the range and size class of the parameters investigated, the following trends are established:

- Both the rotational and vortex noise components are reduced with lower tip speeds.

- At a given tip speed, the rotational noise component is reduced with increased blade loading. For a given tip speed and blade loading, the rotational component is reduced with increased number of blades and reduced thrust.

- The vortex noise component increases with increased thrust and blade loading. The effect of number of blades on the vortex noise is not discernible.

  For the particular rotor configurations investigated, the vortex noise component predominates.

In addition to the parameters varied during the subject test program, it is believed that vortex noise may be affected significantly by the blade twist, taper and section. Different tip shapes and caps have been proposed; however, available evidence indicates only slight gains are possible with these modifications. It is probable that a blade area larger than just the tip is involved. Decreasing the local blade loading over the outboard portion of the radius (as much as 20 per cent) may be quite effective in reducing the vortex noise.

(2) Blade Slap

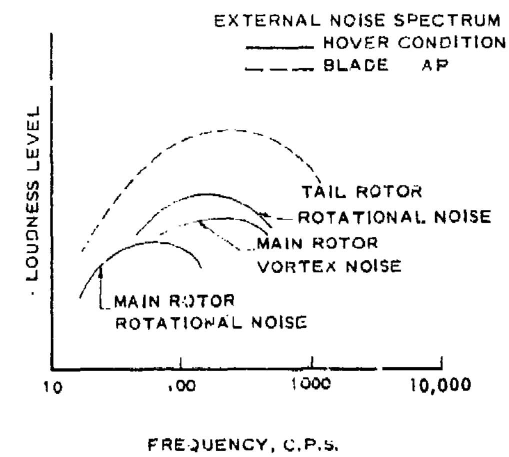
(a) Description and General Discussion - By far, the most objectionable noise associated with helicopter operation, when it occurs, is blade slap. This noise is characterized by its occurrence at blade passage frequency. Depending on its intensity, it has been referred to as a "popping" or "cracking" sound. The term, blade slap, as used herein, denotes one type of noise which may vary in intensity and quality depending on the rotor design and flight conditions.

1) Single Rotor - The single rotor HU-1 helicopter exhibits a tendency to slap under certain flight conditions such as low-power letdowns, decelerations, turns, and moderate forward speeds at high gross weight. The H-13 helicopter also exhibits this characteristic; however, the noise produced by the smaller machine is less severe.

During the tests of the subject program, severe blade slap was not encountered; however, during the fly-over tests of Configuration I, the slap was noted as the helicopter decelerated and turned away from the observer. Narrow bandwidth analyses of these fly-over data reveal that blade slap is composed of all audible frequencies modulated at the characteristic blade passage frequency. An example of this effect is shown by Figure 26, where a large increase in sound pressure level due to slap at a frequency of 500 c.p.s. is shown.

The effect on the over-all frequency spectrum is shown by Figure 27 where the results of a narrow bandwidth analysis of blade slap are given. In this figure, the peak sound pressure levels associated with blade slap are plotted as a function of frequency. It may be noted that blade slap extends over a frequency range of [illegible] to over 1000 c.p.s. and has a maximum intensity around [illegible] c.p.s.

The difference in character between the noise associated with blade slap and the usual helicopter noise is shown in the sketch below. A numerical scale of loudness is not given due to the different conditions of the two tests. Note the broad spectral distribution of blade slap as compared to the hovering frequency distribution of the various noise components. When blade slap occurs, the individual noise sources are not discernible.



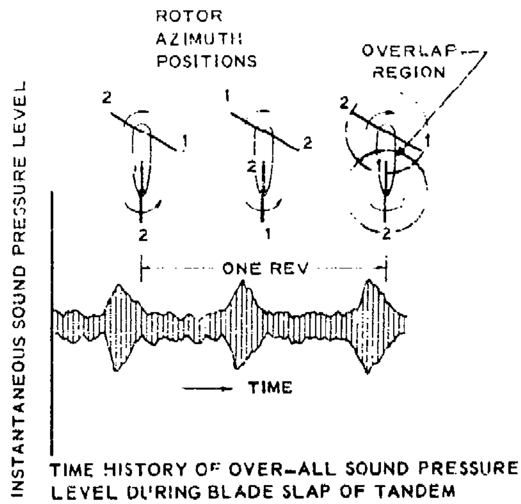
DIFFERENCE IN CHARACTER OF BLADE SLAP AND NORMAL HELICOPTER NOISE

During the test program with the two-bladed HU-1A rotor, blade air load and acoustical measurements were recorded simultaneously during fly-over. Similar data were taken during normal flight with internal noise measurement equipment. Figure 28 gives an example of these data. The recorded differential pressures at various chordwise stations at the 75 per cent radial station are given together with the output from a microphone located in the cabin for a 60-0 knot deceleration when blade slap was present. Note the sharp differential pressure changes occurring at blade azimuth positions of approximately 90 and 270 degrees. The acoustical measurement (trace number 8) also indicates two/rev increase in rotor noise near 90 degrees and 270 degrees azimuth position (a time delay of about 0.02 seconds should be taken into account for the noise to reach the cabin). Sufficient details are not present to define which blade produces the slap. A closer study of the internal noise trace, however, shows a 150 c.p.s. frequency which corresponds to a pressure variation near the blade tip on the advancing side. (See detail in Figure 28.)

2) Tandem Rotor - Based on this contractor's experience with the tandem rotor HSL helicopter and recent discussions with a tandem rotor helicopter manufacturer (Reference 15), it is concluded that the tandem configuration is more susceptible to blade slap than a single rotor configuration. With the twin two-bladed rotors of

the Bell HSL, two/rev blade slap was present during all operating conditions with varying degrees of severity.

Oscillograph records of this noise from Reference 17 show peak sound pressure levels of 129 decibels at a location near the HSL rotor on tiedown. The root-mean-square sound pressure level for the same condition was approximately 118 decibels. The adjacent sketch represents a time history of the over-all sound pressure level during the subject HSL acoustical measurements. Note the position of the rotors. The sudden rise in sound pressure level associated with blade slap is seen to occur periodically at blade passage frequency for one rotor (two/rev). Based on the analysis in Reference 17, the noise originated as the rotor blades entered the overlap region. As in the subject program, sufficient data were not obtained during this work to determine which blade(s) produced the slap.



TIME HISTORY OF OVER-ALL SOUND PRESSURE LEVEL DURING BLADE SLAP OF TANDEM ROTOR HSL HELICOPTER

Based on observations and information from Reference 16, the tandem rotor YHC-1A and HC-1B helicopters generate blade slap in most flight conditions; and, at high forward speeds, a cracking sound is encountered. Indications are that the YHC-1A noise is more intense than the HC-1B at that condition. Although details are not available, it was learned during Reference 16 that preliminary analysis indicates that the blade slap of the HC-1B is generated as the aft rotor leaves the region of overlap. This is in disagreement with the conclusions of Reference 17.

There are many unanswered questions concerning blade slap and this points out the lack of a basic understanding of the origin of this noise and the effects of various parameters on this source. A review of the available literature shows no discussion of this characteristic noise associated with helicopters. The following section presents this contractor's views on the possible origin of blade slap and the approaches to minimize this disturbing noise.

(b) Possible Origin and Mitigation of Blade Slap - Based on the available evidence, the most probable cause of slap at low flight speeds is the rapid angle of attack changes which a blade experiences as it encounters its own, or a previous blade's wake. Compressibility could increase the severity of the effect of this angle of attack change. When the blade angle of attack is suddenly changed, the lift and consequently the trailing wake system also change abruptly. The abruptness of the change in the wake leads to an impluse type noise which contains all frequencies. Less severe angle of attack changes can alter the characteristics of the boundary layer on the blade and the normal vortex noise may be reinforced at blade passage frequency.

This explanation is supported by Figure 29 which shows the results of air load measurements on the HU-1A rotor from Reference 8. That figure shows the differential pressure near the tip of a single rotor helicopter main rotor blade at various forward speeds. It can be seen that a sudden increase and decrease in the blade differential pressure occur during low forward speeds just before an azimuth position of 90 degrees and just after 270 degrees, respectively. At forward speeds of 20 and 40 knots, blade slap was encountered. By observation, this noise occurred at twice per revolution; however, the azimuth position of the rotor blade where slap originated could not be determined.

The possible azimuth positions at which interference with the trailing vortex system might occur are shown in the schematic on Figure 29. The indicated tip vortices would cause sudden inflow changes near azimuth positions A and B. Thus, the passage of a blade through the trailing vortices would result in a sudden force variation on the blade elements near the tip which could result in the noise observed. In all probability the noise would occur on the advancing side.

An explanation for the dependence of the intensity of blade slap on flight condition is obtained by the consideration of the rotor inflow variation for different flight conditions. In a climb where slap is rarely encountered, the rotor's trailing vortex system is directed away from the blades. Conversely, in a partial power descent where blade slap is encountered, the helicopter flies into its wake.

A tandem helicopter is more susceptible to blade slap since the trailing vortex system from two rotors are present. For the tandem, the reasoning presented above remains valid for that machine; and, it becomes possible to encounter blade slap under all operating conditions.

At higher flight speeds where the rotor wake effects are less pronounced (see Figure 29) it is believed that blade slap or crack is of a different origin than that encountered at the lower speeds. For this case it is believed that the severe cracking noise is produced by local shock waves on the advancing blade. These local shock waves, combined with Doppler effects, could explain the high speed cracking noise.

Clearly, additional work is required to define the phenomena associated with blade slap, and to investigate means to reduce it. Based on the available evidence, it is believed that blade slap may be mitigated by decreasing the operating tip speed, the overall blade loading and compressibility effects (airfoil section, special tip shapes, etc.). It may also be possible to reduce blade slap by decreasing the local blade loading at the tip; this could be accomplished by increasing the inboard loading by blade twist or taper. Note that the items theorized to decrease blade slap are the same as those which decrease the vortex noise component.

For the tandem rotor helicopter, blade slap might also be alleviated by minimizing blade overlap and increasing the vertical separation between the rotors' tip path planes. The latter might be accomplished by angular or linear vertical displacement; however, such an approach would not alleviate the problem for all flight conditions (i.e., downwind hovering).

c. Tail Rotor - It was mentioned earlier and is shown by Figure 15 that the noise produced by the tail rotor of the test helicopter is primarily rotational. Peaks in the sound pressure level at multiples of the blade passage frequency were the only significant tail rotor noise measured. By reference to Figure 18, it is seen that the loudness level of the tail rotor is significantly greater than that of either the rotational or the vortex component of main rotor noise.

Decreasing the tip speed and increasing the number of blades are theoretically effective means of reducing rotational noise. The effect of tip speed is illustrated by Figure 30 which shows that as the tip speed is reduced, the loudness level of all harmonics of the rotational component are reduced. Another advantage of lower tip speed operation is also shown. Note the reduction in loudness level at the higher frequencies associated with low tip speed operation. This is believed to be an r.p.m. effect on the higher harmonic loudness level.

Since the scope of the subject test program did not include variations in tail rotor parameters, existing propeller theory and certain assumptions based on this test program are used to indicate the possibilities of tail rotor noise reduction. It is believed that by this approach, trends can be established to evaluate the effects of number of blades, tip speed, thrust, etc., on the perceived loudness of tail rotor noise.

Figure 31 gives the calculated sound pressure level for the rotational and vortex components of a tail rotor with two, three and four HU-1A tail rotor blades. These calculations were based on the HU-1A tail rotor hovering thrust and power requirements; constant power and thrust were assumed. For reference, a line of constant rotor thrust coefficient/solidity is also given ($C_T/\sigma = .05$). Figure 31a shows that the magnitude of the fundamental rotational component is reduced by lowering the tip speed and increasing the number of blades. The corresponding measured data for the HU-1A tail rotor are also given. It is seen that although theory predicts a rotational sound pressure level significantly lower than that measured, the general trend with tip speed is valid. Comparisons of the calculated higher harmonics of the rotational component show that these sound pressure levels are underestimated even more than the fundamental. Fortunately, it is indicated by the tests of this program that the magnitude of all harmonics of the rotational component will be reduced if the fundamental is decreased. Therefore, calculations of the fundamental rotational noise are used herein to provide an indication of the noise reduction trends as a function of the tail rotor aerodynamic parameters.

The calculated maximum vortex sound pressure levels for the above tail rotors are shown in Figure 31b. The reference $C_T/\sigma = .05$ is also given. The calculations show that the vortex noise increases with number of blades (the chord is held constant), and with higher tip speeds. Above tip speeds of about 600 f.p.s. the vortex noise is shown to be less than that of the rotational component. This is also indicated by the test results of this program. The calculated data indicate also that for certain combinations of design parameters the rotational and vortex noise will become of equal importance.

It is also seen from Figure 31 that for a given basic design the performance requirements define the noise reduction possibilities for the tail rotor. Based on the use of the HU-1 tail rotor blade, a tip speed limit of about 450 f.p.s. is apparent at a value of C_T/σ of .05. Below that tip speed, the vortex noise will predominate due to the increased blade area required to maintain the design thrust factor.

To evaluate the theoretical results of Figure 31 in terms of loudness level, the magnitude of the higher harmonic rotational noise and the frequency of the peak vortex noise must be established. Using the test data of the subject program as a guide, the following assumptions are made to enable this transformation.

1) The magnitude of the nth harmonic of the tail rotor rotational sound pressure level may be estimated by reducing the calculated sound pressure of the fundamental by an amount based on the difference in magnitude between the fundamental and nth harmonic measured during this program.

2) The frequency of the peak vortex noise of the HU-1 tail rotor blade is approximately twice that of the HU-1A main rotor, or 1000 c.p.s. at 6400 engine r.p.m. This is based on the equation $f = K \frac{V}{d}$. Since V_t for the tail and main rotors is nearly the same, the frequency ratio for the main and tail rotors is approximately equal to the ratio of their respective chords.

It is believed that these assumptions are valid for the HU-1 and for helicopters of the same general size class operating at similar rotational speeds.

The results of this transfer of the theoretical sound pressure level to loudness level are shown by Figure 32. That figure shows the calculated loudness level of the fundamental and peak rotational components and the vortex noise as a function of tip speed and number of HU-1A blades. Measured peak rotational loudness levels of the HU-1A tail rotor are also shown. The minimum values of tip speed as a function of C_T/σ are also indicated.

It is seen that a single curve represents all numbers of blades for the fundamental rotational noise. This is due to the transfer from sound pressure level to loudness level and the interaction between rotor speed and blade loading. The transfer from sound pressure level to loudness level is dependent on the fundamental or blade passage frequency and must be evaluated for each case. The changing slopes and intersections of the curves representing the number of blades are also due to that transfer.

Figure 32 also shows that for the cases under investigation, rotational noise is predominant above tip speeds of about 600 f.p.s. Additionally, note that the test data of the subject program show peak rotational loudness levels about 10 phons higher than the estimated values. For purposes of direct comparison, estimated tail rotor loudness levels will be so modified in Section VI.

In addition to reducing tip speed and blade loading, other possibilities with respect to the reduction of tail rotor noise include twist and taper to minimize the outboard blade loading and unloading the tail rotor at high speed by a tail fin. The location of the tail rotor with respect to the main rotor and tail boom may also be important. This could not be verified conclusively during this program; however, there were indications that interference and masking effects were present which influence the tail rotor noise.

2. Drive System

The noise generated by a helicopter drive system consists of that associated with the transmission(s), couplings, bearing supports, and drive system vibrations. These sources contribute substantially to the internal noise characteristics of the helicopter but have little effect on the far field external level. This is due to the atmospheric attenuation of high frequency noise and the masking effects of other sources.

From Reference 18, the noise caused by the operation of a gear is a result of stress waves produced in the gears, air and oil pocketing, friction, impact, and the variation of radial forces. Proper gear geometry and accuracy of gear manufacturing processes reduce the level of all of these noise sources; however, the noise due to friction remains a function of the torque input. The sudden reversal of the frictional forces on each tooth, gives rise to an effect called "pitch line shock" and results in a pronounced noise generated at the tooth-contact frequency.

The contribution of the drive system sources and accessories to the internal noise spectrum of the HU-1A helicopter is shown by Figure 33. The prominent high frequency noise peaks can be traced to the main transmission generator and the first and second stage planetary pinions (gears) of the main transmission. These peaks are identified by calculations of the output frequencies and the gear-tooth contact frequencies of these sources, respectively.

In addition to gear noise, the noise associated with bearing supports, couplings, and drive system vibrations may become significant in a tandem rotor helicopter due to the relatively long interconnecting shaft between the two main rotors. Since the shaft is normally located above the passenger compartment, this area may be subjected to a considerable amount of high frequency noise, both air- and structure-borne.

Theoretical analyses are not available to predict the effect of various noise control techniques in reducing gear noise in a helicopter drive system. Several techniques have been suggested. These include helical gearing, elastically mounted ring gears, plastic gears, increased transmission housing thickness, and various substitutes for gears. Except for severe problem areas, it is doubtful that the noise reduction achieved by these techniques would warrant the necessary changes to an existing design.

Lightweight material such as fiberglass, foam, etc., is often used in existing designs to decrease the noise transmitted from the source to an observer in the vehicle. In addition, damping tape is used extensively where structure-borne sound is transmitted through long fuselages and structures. Test work to evaluate the noise reduction that can be obtained by incorporating the above techniques in future helicopter drive system designs is needed.

3. Power Plant

a. Turboshaft Engine - The noise of a turboshaft engine is associated with: (1) the inlet (primarily compressor whine), (2) the engine drive system, gearing and bearing noise associated with the reduction gears and the accessory drives, (3) the exhaust, including the noise produced in the mixing region of the exhaust gas stream and the surrounding air, plus the contributions from combustion and the power turbine, and (4) the noise radiated from the structural vibrations of the engine case. The major sources of a typical turboshaft engine installation are the compressor whine and the exhaust noise.

Compressor whine is associated with the disturbances caused by the passage of air by the compressor blades, similar in nature to the mechanism of rotational noise as discussed previously. The frequency of the compressor noise is determined by the number of compressor blades, the number of stationary blades and the rotational speed. The noise from the first compressor stage is normally the predominant source; although, for certain designs the noise from the following stages may become noticeable. These stages generally have different numbers of blades and therefore different fundamental frequencies. Usually, the frequencies associated with compressor noise of a turboshaft engine are around 10 kilocycles and are quite directional. This noise is attenuated rapidly with distance; thus, only the internal and near field levels are significantly influenced by this source. Although detected in the cabin of the HU-1A helicopter, the frequency response of common acoustical instrumentation is such that this source is not easily identifiable from test data.

The exhaust of a turboshaft engine is not a powerful noise source when compared with the helicopter main and tail rotor. The exit velocity of the exhaust gases is relatively low (300 feet per second at the exit nozzle for the T53-L-1A engine); thus, the turbulence and the noise of the exhaust mixing region are relatively small.

From the standpoint of external noise, the turboshaft power plant offers distinct advantage over other engines for rotary wing aircraft. Due to low velocity exhaust gases, the aerodynamic noise produced by the exhaust is below that of the pure jet, tip jet, ducted fan, etc. Although influencing the near field noise, the high frequency turbine and compressor noise is fortunately attenuated rapidly with distance.

The use of commercially available acoustical material internally offers the most reliable method of attenuating high frequency drive system and engine noise. Incorporating such material in the cabin area of an existing design would require additional treatment to any acoustically weak areas, such as windows, door seals, access ports, electrical outlets, etc.

Treatment of the interior engine and transmission cowling enclosure with acoustical material would present a large absorption area for high frequency noise; however, the design would have to be such as to avoid an engine and/or transmission cooling problem. A more complete discussion of internal noise is presented in Section V-E-5.

b. Piston Engine - The primary source of noise of a piston engine is the exhaust. This noise originates from the periodic expulsion of hot gases of combustion through the exhaust system. This represents a periodically changing volume which by definition is an elementary noise source. The lowest frequency of the exhaust noise spectrum usually corresponds to the firing frequency of the engine. Harmonics of this frequency may also be noticeable. Based on observations and data reported in Reference 4, the exhaust noise is the major internal and external noise source of piston engine powered helicopters.

Exhaust mufflers offer the possibility of noise reduction for reciprocating engines although the weight and performance penalty may often be critical. These penalities are also a factor in considering other possible noise reduction techniques such as combining the exhaust of several cylinders into one exhaust port to partially cancel some of the components of the exhaust pressures, and having pairs of cylinders working in counterphase to cancel their fundamental firing frequency.

E. RELATIVE PROMINENCE

In previous sections, the major noise sources of a helicopter have been identified in relation to the helicopter component from which they emanate and the mechanism by which they are produced. In this section, the relative prominence of these noise sources and components is given for the far field case and discussed in relation to the position of the observer with respect to the source and the terrain. Additionally, the internal noise levels of the helicopter are discussed.

1. Far Field Case

Based on the investigations of this program, the noise components of a turbine powered helicopter in their order of prominence for the far field case (200 feet or more) are:

1) Main rotor blade slap (when it occurs).

2) Tail rotor (rotational).

3),4) Main rotor (vortex, rotational).

5) Drive system.

6) Power plant.

For piston engine helicopters, power plant exhaust noise becomes a predominant source, second only to blade slap.

2. Effects of Position

The position of the observer in relation to the sources influences the relative prominence of the noise components perceived. The importance of the directivity pattern associated with certain sources has been discussed previously (Figures 11 and 12, over-all sound pressure level). For the subject study, it must be assumed that an observer may be at any angle with respect to the helicopter; consequently, directions of maximum sound pressure have been used.

The distance of the observer from the helicopter is important in defining the relative prominence of the various noise components. The principal effect of distance is to attenuate the high frequency sounds. Figure 34 (from Reference 19) shows the atmospheric attenuation coefficient, k, in decibels/1000 feet, as a function of frequency. At typical helicopter transmission frequencies of about 1200 c.p.s., the sound is reduced 4 decibels per 1000 feet. It is seen from this, that transmission and other high frequency noise may be considerably more prominent inside the helicopter than indicated by the far field noise measurements of this program.

3. Effect of Terrain

In low altitude flying, the terrain affects the noise propagated between the helicopter and the observer by absorbing and reflecting a portion of the sound. Obviously, in military applications the protection offered by hills and other protrusions between the helicopter and an enemy could be used wherever possible to decrease the distance of both sight and sound detection. A significant amount of sound absorption due to thick land vegetation, such as wooded and brush areas, is achieved, but only at low elevation angles as shown in Figure 35 (taken from Reference 19). In Figure 35, the propagation loss coefficient, k, represents the reduction in noise level per 1000 feet distance between the noise source and observer due only to the absorption properties of the terrain. This term does not include the attenuation of high frequency noise due to the atmosphere. As can be noted, a zero to ten decibel loss per 1000 feet can be achieved in partly to heavily wooded areas at low elevation angles.

4. Effect of Flight Technique

There may be many military and civilian situations where it is desirable to minimize the far field noise of existing helicopters. For these cases, the helicopter should be operated at as low a tip speed as practical. Steady cruising near minimum power will minimize the possibility of detection or annoyance; high speeds should be avoided; decelerations to hover should be accomplished rapidly to reduce the time during which blade slap can occur; also, descents in complete autorotation are preferable to partial power descents to minimize blade slap. Autorotation and climbs are not normally associated with blade slap but should be moderate, to minimize the main and tail rotor rotational and vortex noise. It is also desirable to fly as low as possible.

It is realized that some of the above techniques are conflicting; for instance, low tip speed - low altitude operations are not necessarily compatible. The techniques are noted, however, since during extreme circumstances it will be at the discretion of the pilots of the helicopters to use all possible techniques to the maximum extent possible.

5. Internal Noise

The internal noise characteristics of helicopters are even more complex than those for the far field case. This is not only due to the relative position of the observer, but also due to the fact that structural as well as airborne noise is involved. Further, the criteria for internal noise reduction are normally more severe due to requirements to minimize annoyance, fatigue, speech interference, etc.

A typical frequency spectrum of the noise in the cabin of the single rotor HU-1A helicopter is given by Figure 35. As discussed previously, the low frequency peaks in the 10- to 100-c.p.s. frequency range are associated with the main rotor rotational noise. The maximum sound pressure levels correspond to the harmonics of the main rotor rotational speed, 4/rev, 6/rev, 8/rev. Tail rotor rotational noise and transmission gear noise are present in the mid-frequency range at somewhat lower sound pressure levels.

Figure 36 shows the loudness level of the major HU-1A internal noise components with the pilot and copilot windows opened. It is seen that the internal transmission noise has approximately the same loudness level as the main and tail rotor components. With the cockpit windows closed, the main and tail rotor noise levels are reduced from 3 to 6 decibels with the equivalent reduction in loudness level. For this condition, the most prominent internal noise is that of the main transmission and accessory generator.

Figure 37 shows the possible sound pressure level reduction for the HU-1 for a proposed acoustical treatment given in Reference 20. Note the significant sound pressure level reduction at the higher frequencies for both the hover and cruise conditions.

All helicopter configuration changes made to reduce the far field noise will result in a reduction in the internal noise level. The external noise levels may be used to some extent to evaluate the relative effects of configuration changes on internal noise by considering the atmospheric attenuation discussed previously.

6. The Concept of Acoustically Balanced Design

The use of loudness level as a criterion for evaluating the prominent noise sources brings about the introduction of the concept of the acoustically balanced design. For an optimum noise reduction design with respect to aural detection, observer reaction, etc., all components should be perceived equally. It may be impractical to design for such a case; however, such a consideration can define specific areas which should first be attacked. In attempting to create an acoustically balanced design for the HU-1, the tail rotor noise should first be attacked, followed by the main rotor vortex noise, etc. Blade slap is not in this listing for the HU-1 since it is believed that an attack on this problem should be approached as a research effort.

VI. EVALUATION

In this section the results of this program are used to evaluate the penalties associated with reducing the noise level of the HU-1 helicopters. Modifications to reduce the noise level are presented and evaluated in conjunction with considerations of cost, performance and weight. Recommended configurations are given and compared with the production machines. The comparisons presented in this section are for similar mission conditions; that is, where the helicopters have the same crew load, payload, and have full fuel at take-off.

A. TAIL ROTOR

In Section V it was shown that except when main rotor blade slap occurs, the most prominent noise associated with the HU-1 helicopters is that produced by the tail rotor. For the HU-1, blade slap may be partially mitigated by operational techniques; for the general case, additional work is required to define and eliminate that source. For these reasons, consideration is first given herein to the reduction of tail rotor noise.

Figure 32 shows the estimated loudness levels of several HU-1 size tail rotors as a function of blade tip speed. As noted, for tip speeds greater than about 575 f.p.s., the rotational noise component is the principal noise. In all cases, it is desirable to reduce the tip speed as much as possible.

The extent to which the tail rotor tip speed can be reduced depends on considerations of the performance requirements of the helicopter (V_{max}, maneuverability, altitude performance, etc.) as well as those of weight and center of gravity. From the standpoint of performance, the major design parameters are the blade loading and tip speed. These items define a mean blade lift coefficient or C_T/σ, from which altitude, maneuverability, and forward speed stall limits are established. For the parameters of the HU-1A tail rotor blade, the minimum tip speeds associated with several values of C_T/σ are shown by Figure 32. These minimum tip speeds are related to the large increase in blade area needed to maintain the values of C_T/σ and will vary depending on the size of the blade being considered. By combining these approximations of the noise level and the limiting velocities associated with blade stall, an indication of the minimum tail rotor noise level as a function of the stall limited design speed is obtained.

Figure 38 shows such a plot and also a calculated stall limit speed and hovering noise level for the HU-1A tail rotor. The shaded area represents variations due to stall angle, effects of number of blades, etc. It is seen that as the stall limited design speed is increased from 0 to about 80 knots, the tail rotor loudness level does not increase appreciably. For helicopter design speeds above 80 knots, the reference loudness level increases significantly (approximately 55 to 80 phons from 120 to 180 knots). This points out a possible problem area with high performance helicopters; however, the solution of unloading the tail rotor in forward flight is immediately suggested. This is illustrated on the figure.

From Figure 32, it is seen that for the HU-1A tail rotor, the rotational noise component predominates. Consequently, a significant noise reduction may be expected by decreasing the tip speed. The effect of number of blades depends on the particular blade area-tip speed combination. The extent to which the tip speed may be reduced is illustrated by Figure 39 where tip speed is shown as a function of blade loading for several values of the stall limit speed. Also shown are the blade loadings for tail rotors with 2, 3 and 4 production HU-1 blades, and a practical tail rotor tip speed ratio limit based on considerations of flapping and fatigue loads. It is seen that from these considerations a tail rotor tip speed of about 590 f.p.s. is required if it is desired to maintain the 150 knot stall limited speed of the HU-1A. Because of such effects as maneuverability at altitude, low engine speed operation, etc., maintaining that maximum allowable speed is desirable and the minimum practical tip speed for the HU-1 tail rotor is defined as 590 f.p.s.

From Figure 39, adding one and two blades to the HU-1A tail rotor reduces the minimum acceptable tip speed to about 630 and 600 f.p.s., respectively. Adding additional blades would result in violating the tip speed ratio limit. It is seen, however, that a slight additional reduction in tip speed may be achieved by a small reduction in blade loading (or chord increase).

The over-all effects of these tip speed reductions and the related changes are given in Table 7. The loudness levels were obtained from Figure 32. The "calculated" values of the rotational noise component are increased 10 phons to account for the discrepancy between theory and the experimental data of this program. The principal reasons for this discrepancy are believed to be due to such items as main rotor interference and assumed blade force distribution (total thrust located at three-quarter radius).

Although from the performance standpoint it is desirable to maintain the maximum rotor diameter possible, small changes in diameter would be acceptable. Therefore, it would be possible to reduce the diameters of the rotors shown in the table to allow unrestricted operation during test without a tail rotor gear box change.

Over-all power requirements due to the above tail rotor modifications are negligible. The relative weights of these tail rotors would change approximately as shown by the table; however, the actual weights would be dependent upon the detail design and may vary considerably from the indicated values. Even though the weight of these components may change by a factor of two, the net effect on the helicopter's empty weight is small. For this presentation, it was assumed that the existing center of gravity could be maintained by items of fixed equipment.

It should be remembered that the data of Table 7 are intended only to indicate the numerical values associated with the trends. As mentioned before, the state of the art is not sufficiently advanced to define quantitatively the noise level of rotors. Within this frame of reference, Modification b is selected as the optimum tail rotor for the thrust and stall limited speed requirements investigated. Its loudness level is only one phon higher than the minimum noise of Modification d; however, production blades may be used and the system should be lighter. A new gear box and hub are required. Modification b will result in a loudness reduction of about 50 per cent of that of the standard HU-1 tail rotor.

B. MAIN ROTOR

In this section the main rotor acoustical data of this program are used with standard performance analysis techniques to investigate the effects of various noise reduction modifications on the HU-1 main rotor. This is accomplished by summarizing all of the main rotor acoustical data of the program, noting general trends, and then evaluating selective modifications to the HU-1A to determine their total effects. Practical rotor configurations are investigated in all cases, and emphasis is placed on maximum use of existing HU-1 components.

On the basis of the test results of the subject program, the trends of the main rotor hovering loudness levels versus blade loading for two-and three-bladed designs and for two tip speeds are shown in Figure 40. The loudness levels shown are based on the peak values of the rotational and vortex noise as illustrated by Figures 24 and 25, and are used herein to estimate main rotor noise as a function of the various rotor parameters.

The influence on performance for the various main rotor design parameters which were selected on the basis of the noise reduction trends is given in Table 8. Hovering ceiling, range, maximum rate of climb, and maximum stall limited speed, as well as the loudness level, are shown for the HU-1A and various modifications of that machine. For the performance data, a crew of two, full fuel, and a typical payload (1000 pounds) are used. Empty weights are established on the basis of the three rotor configurations tested. Since the three-bladed rotor of this program is experimental, the weight shown for that rotor system is somewhat high in relation to a production design.

Modification 1 involves the use of two 21-inch chord blades on the HU-1A operating at low engine r.p.m. It is seen that the hovering loudness level is appreciably lower than that of the reference HU-1A at normal rotor speed and the performance of the machine is improved.

Modification 2 (three 21-inch chord blades) results in a vehicle with the lowest loudness level in hover; however, this configuration results in an unacceptable decrease in performance. This is due principally to the increased weight associated with the larger rotor; that is, there is too much blade area for the installed power of the HU-1A.

Modification 3 consists of three 15-inch chord blades and is found from the noise standpoint to be equivalent to Modification 2. This configuration compares unfavorably from a performance standpoint, however, because of the higher rotor weight.

With added fuel capacity and a more powerful engine, blade area can be used to advantage. Both increased fuel capacity and a more powerful engine are provided by the HU-1B helicopter. Data for that machine are shown in the table. It is seen that the performance items of the HU-1B are considerably improved over those of the HU-1A. However, the loudness level of the HU-1B main rotor is slightly greater than that of the HU-1A.

Modification 4 involves the use of three 21-inch chord blades on the HU-1B. This modification is shown to be appreciably quieter than the standard HU-1B; however, the over-all performance is decreased. The loudness level of Modification 4 is not as low as that of Modification 2 (same rotor system on the HU-1A) because of the higher gross weight. The noise level of the three 21-inch chord blades on the HU-1B is shown to be about equal to that of the HU-1A with two 21-inch chord blades (Modification 1).

Modification 5 shows the effect of increasing the chord of a two-bladed rotor to add sufficient blade area to minimize rotor vortex noise. It is seen that two 27-inch chord blades produce slightly more noise than the three-bladed rotor (Modification 4); however, the performance of the two-bladed rotor is slightly higher (than Modification 4) because of its lower gross weight.

Finally, a very wide chord two-bladed rotor is shown as Modification

6 (two 31.5-inch chord blades). The performance is slightly better than for Modification 5; however the loudness level is increased. This is the only main rotor configuration which is found to have a predominant rotational noise component.

The reduction of tip speed for all modifications discussed thus far is accomplished by lowering the engine speed from 6400 to 5800 r.p.m. Further reduction of engine r.p.m. would be impractical since the loss of available engine power would be unacceptable. Modification of the HU-1 main rotor transmission to provide lower tip speeds for the purpose of noise reduction <u>only</u> is not considered to be justified.

The remaining method of lowering the tip speed is to reduce the rotor diameter. To evaluate this, it will be assumed that the influence of diameter on the loudness level can be determined by extrapolating data from Figure 40, through changes in tip speed and blade loading. For small variations in diameter, it is believed that this assumption will not mask the trends associated with the changes. For large diameter changes where major rotor frequency shifts are involved due to necessary changes in design rotor r.p.m., the loudness level data in Section V must be converted to sound pressure level and re-evaluated on the basis of the new frequency spectrum. Table 9 gives several examples of the effects of diameter.

It is shown that a reduction of rotor diameter will decrease the loudness level. For the cases shown, however, a reduction of both stall limit speed and hovering ceiling results and is considered to be unacceptable.

Because of uncertainties in using the data of Figure 40 for other than 44-foot diameter rotors, additional tests and analyses are needed before definite conclusions can be drawn. The data of Table 9 indicate, however, that significant loudness level reductions are possible with small diameter - large chord rotors operating at low tip speeds.

C. DRIVE SYSTEM AND POWER PLANT

In comparison with the main and tail rotor, the external noise generated by the engine and the transmission systems of the HU-1 helicopter is considered to be negligible. No modifications are considered.

D. TOTAL EFFECTS OF MAIN AND TAIL ROTOR MODIFICATIONS

In this section the results of the application of the noise reduction techniques studied during this program are summarized by defining the optimum configuration for the HU-1A and HU-1B where noise, performance and costs are considered. These recommendations and comments are intended to be a guide only. As mentioned earlier, the state of the art of noise prediction and control is not sufficiently advanced to define the rotor noise characteristics accurately. Further, the numerical values assigned to the various characteristics are valid only for the mission selected. In reviewing this work, consideration must be given to these items.

In connection with the modifications to the HU-1A and HU-1B, items which are considered but for which no numerical values were assigned include autorotation and flare characteristics, cockpit vibrations, and rotor fatigue loads. It is assumed that for the gross weights shown, pilots of helicopters with the modifications could operate at low engine speed in situations where low noise level is required; in other flight situations (at altitude, high speed, etc.) full r.p.m. could be used.

Table 10 summarizes the best configurations studied during this program (from Tables 7 and 8). In this table an attempt is made to represent the total effect of the changes as a percentage. It is realized that adding noise of the type considered on a loudness scale is questionable. The results of such an approach, however, agree with qualitative observations of existing configurations and are therefore included. For this, two approaches are taken: 1) to define the percentage reduction in loudness (sones) for the component with the highest loudness level (for this case, the tail rotor), and 2) to convert the peak loudness level of all noise components to sones, and add to obtain the total loudness.

Table 10 shows that the tail rotor loudness can be reduced 50 per cent by Modification b (four-bladed, 8.4-inch chord rotor, V_t = 600 f.p.s.). To realize this reduction, however, it is necessary to operate the main rotor at low tip speed. For unrestricted helicopter operation, it is necessary to provide added blade chord for these low tip speeds. If the HU-1A main rotor tip speed is maintained, main rotor noise will mask the effects of the new tail rotor and only about a 20 per cent noise reduction will be realized.

When the effects of other main rotor modifications are included, the situation changes somewhat, although it is still apparent that the tail rotor modification is the most significant. Referring to Table 10, the lowest total loudness is indicated for low tip speed operation of the HU-1A, modified with a two-bladed, 21-inch chord main rotor and a four-bladed tail rotor. With these modifications, a reduction in total loudness of approximately 40 per cent can be realized. It should be noted that the performance of this helicopter is shown to be increased slightly over that of the basic HU-1A, operating at normal tip speed.

From the standpoint of total loudness, the power plant and fuel capacity of the helicopter contribute significantly toward defining the optimum noise reduction modification. For this reason, the basic HU-1B and several modifications of that machine are given. Two HU-1B modifications are shown in Table 10 since the performance and noise of the two are approximately the same. In addition to the four-bladed tail rotor, the modifications considered include a three-bladed, 21-inch chord and a two-bladed, 27-inch chord main rotor. It is seen that the loudness of the three-bladed rotor is slightly lower; however, the performance of the two-bladed rotor is slightly superior. On the basis of the performance advantage, the wide-chord two-bladed rotor configuration is selected as the optimum. With this main rotor modification and the four-bladed tail rotor, a reduction of about 40 per cent in total loudness of the HU-1B can be realized.

It should be noted that for all cases the principal noise reduction resulted from the new tail rotor. Also, the noise reduction achieved for the main rotor resulted from lower tip speed operation. Because of the low tip speed, the blade loading had to be reduced to maintain the proper high speed, maneuverability and altitude performance (C_T/σ).

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TABLES

TABLE 1

TEST ROTOR CONFIGURATIONS

	Number of Blades	Diameter Feet	Disc Area Sq Ft	Blade Area Sq Ft	Solidity	Airfoil/Chord (Constant over Radius) NACA/Inches	Blade Twist Degrees	Tip Speed Ft/Sec Min	Tip Speed Ft/Sec Normal
Main Rotor Configuration									
I	2	43.75	1503	55.0	.0368	0015/15.2	-12	652	720
II	2	44.0	1520	77.0	.0506	0012/21.0	-10	656	724
III	3	44.0	1520	115.3	.0760	0012/21.0	-10	656	724
Tail Rotor Configuration									
Standard	2	8.4	56	5.9	.105	0015/8.41	0	643	710

TABLE 2

TEST SCHEDULE AND HELICOPTER OPERATING CONDITIONS

Test Number	Configuration	No.	Operating Condition	Engine Speed RPM	Gross Weight Lbs	Engine Power HP	Forward Speed Knots
1	2-Bladed	I	Tiedown	5800	--	500	-
2	15-inch Chord		Normal	6400	--	500	-
3				6700	--	500	-
4				Engine Shut Down		-	-
5*				5800	--	500	-
6				6400	--	500	-
7				6700	--	500	-
8				Engine Shut Down		-	-
9			Tail Rotor	6400	--	500	-
10			Disconnected	5800	--	500	-
11				6400	--	500	-
12				6700	--	500	-
13				Engine Shut Down		-	-
14			Hover	6400	6800	-	-
15				6700	6800	-	-
16				6400	5800	-	-
17				6700	5800	-	-
18	2-Bladed	II	Hover	6400	6800	-	-
19	21-inch Chord			6400	5800	-	-
20				6700	5800	-	-
21			Fly-over	6400	6800	-	60
22				6400	6800	-	80
23				6400	6800	-	105
24				6400	6200	-	60
25				6700	6200	-	60
26				6400	6200	-	80
27				6700	6200	-	80
28				6400	6200	-	105
29				6700	6200	-	105
29A**				6400	6200	-	Maneuver -
29B				6400	6200	-	Right Turn
29C				6700	6200	-	
29D				6700	6200	-	
29E				6400	6200	-	
29F				6400	6200	-	
29G				6400	6200	-	
29H				6400	6200	-	
30	2-Bladed	I	Fly-over	6400	6400	-	60
31	15-inch Chord			6400	6400	-	80

27				6700	6200	–	80
34				6600	6200	–	[illegible]
35				6400	6200	–	[illegible]
36				6600	6200	–	[illegible]
27				6700	6200	–	[illegible]
28				6400	6200	–	105
29				6700	6200	–	1[illegible]5
29A**				6400	6200	–	Maneuver -
29B				6400	6200	–	Right Turn
29C				6700	6200	–	
29D				6700	6200	–	
29E				6400	6200	–	
29F				6400	6200	–	
29G				6400	6200	–	
29H				6400	6200	–	
30	2-Bladed	I	Fly-over	6400	6400	–	60
31	15-inch Chord			6400	6400	–	80
32				6400	6400	–	90
33				6400	6200	–	60
34				6600	6200	–	60
35				6400	6200	–	80
36				6600	6200	–	80
37				6400	6200	–	90
38				6600	6200	–	90
39				6400	6200	–	Low power
39A				6400	6200	–	letdown
40				6400	6200	–	Right turn
40A				6400	6200	–	
41				6400	6200	–	Deceleration
42				6400	6200	–	
43	3-Bladed	III	Fly-over	6400	6400	–	60
44	21-inch Chord			5800	6400	–	60
45				6400	6400	–	80
46				5800	6400	–	80
47				6400	6400	–	100
48				5800	6400	–	100
49			Hover	6600	6400	–	–
50				6400	6350	–	–
51				5800	6300	–	–
52				6600	6100	–	–
53				6400	6075	–	–
54				5800	6050	–	–
SR1				6400	6025	–	–
SR2				6400	6025	–	–
SR3				6400	6025	–	–
SR4				6400	6025	–	–
SR5				6400	6025	–	–
SR6				6400	6025	–	–

* Tests 5 through 9, test vehicle turned 180°
** Tests 29A through 29H and 39 through 42, flights conducted for blade slap

TABLE 3
ENVIRONMENTAL CONDITIONS DURING TESTS

Configuration Number	Operating Conditions	Date	Wind Direction-Velocity, Knots	Temperature °F	Humidi %
I	Fly-over	9-27-61	S 14	84.2	52
	Tiedown (Tests 1-8)	10-3-61	ENE 6	66.9	44
	Tiedown (Tests 9-13) and Hover	10-4-61	S 12	69.1	47
II	Fly-over	9-21-61	SSW 22(to 32)	80.7	64
	Hover	10-4-61	S 12	69.1	47
III	Fly-over and Hover	10-31-61	WSW 8	79.3	66

TABLE 4
DATA REDUCTION SCHEDULE

TEST NUMBER	TEST DATE	DATA ANALYSIS		
		OVER-ALL	1/3-OCTAVE	CONSTANT BANDWIDTH FILTER(6-C.P.S.
1	10-3-61	All Mikes	-	Mikes -2, 9, 16, 22
2			-	All Mikes
3			-	Mikes - 2, 9, 16, 22
4			-	-
5			-	-
6			-	All Mikes
7			-	-
8			-	-
9	10-4-61		-	All Mikes
10			-	Mikes - 2, 9, 16
11			-	All Mikes
12			-	Mikes - 2, 9, 16
13			-	-
14			-	Mikes -2, 9, 16, 22
15			-	-
16			All Mikes	All Mikes
17			-	-
18			-	Mikes - 2, 9, 16, 22
19			All Mikes	All Mikes
20			-	-
21-29H	9-21-61		-	-
30-42	9-27-61		-	-
43-48	10-31-61		-	-
49-52			-	Mikes - 2, 6, 9, 13, 16, 20
53			All Mikes	All Mikes
54			-	Mikes - 2, 6, 9, 13, 16, 20
SR1-SR6			-	One Mike

TABLE 5

OVER-ALL SOUND PRESSURE LEVELS FOR TIEDOWN AND HOVER TESTS

Mic No.	Test No.																					
	1	2	3	5	6	7	9	10	11	12	14	15	16	17	18	19	20	49	50	51	[illegible]	53	[illegible]
1	87	92	92	84	88	91	85	81	87	87	90	93	89	93	97	90	91	[illegible]	89	[illegible]	[illegible]	[illegible]	[illegible]
2	88	91	91	83	85	83	85	81	87	86	93	95	91	93	95	91	93	92	92	88	92	91	[illegible]
3	90	92	90	<u>84*</u>	88	<u>93*</u>	83	82	86	89	87	92	92	94	97	92	97	91	[illegible]	87	[illegible]	[illegible]	[illegible]
4	88	88	88	88	90	93	91	88	91	90	92	97	93	94	98	92	95	90	[illegible]	87	[illegible]	[illegible]	97
5	89	90	89	91	92	96	91	89	93	92	94	94	93	93	97	93	94	91	89	86	[illegible]	[illegible]	97
6	89	90	90	87	89	92	87	86	91	90	94	94	92	92	94	91	94	90	89	85	[illegible]	[illegible]	[illegible]
7	88	91	94	86	90	92	92	90	92	90	95	95	94	94	94	94	95	91	91	[illegible]	[illegible]	[illegible]	[illegible]
8	105	105	106	104	103	105	98	101	101	103	103	101	101	103	113	108	107	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]
9	105	101	102	96	100	98	101	103	102	101	104	106	102	103	106	103	103	102	101	100	[illegible]	103	98
10	103	103	104	96	98	99	102	103	104	102	109	108	107	107	109	107	108	101	100	[illegible]	101	101	101
11	99	98	98	97	100	100	102	102	102	101	106	109	104	105	107	105	106	99	99	[illegible]	100	[illegible]	96
12	-	-	-	102	103	102	105	102	101	101	102	108	104	103	104	102	107	[illegible]	97	[illegible]	[illegible]	[illegible]	96
13	97	101	101	99	101	102	103	97	102	100	102	102	101	104	103	103	102	[illegible]	101	[illegible]	[illegible]	[illegible]	95
14	103	102	102	105	105	[illegible]	104	101	102	101	104	104	105	103	103	106	105	[illegible]	[illegible]	[illegible]	[illegible]	[illegible]	97
15	113	112	111	106	104	105	105	105	107	103	112	115	111	112	115	111	113	105	104	103	105	104	103
16	109	110	108	104	106	106	108	108	108	105	111	111	118	109	111	108	109	108	107	107	[illegible]	107	105
17	108	109	108	106	108	106	106	109	108	108	113	113	112	112	115	111	113	107	106	106	108	107	106
18	107	106	107	106	108	106	105	-	-	-	112	113	110	111	112	109	108	105	104	103	106	105	103
19	105	105	105	108	109	108	109	105	105	104	107	107	107	107	107	107	107	105	103	102	104	105	103
20	103	104	103	105	106	107	108	105	105	104	108	107	106	108	107	106	106	103	104	100	104	102	100
21	103	102	102	-	-	-	102	102	105	103	103	106	102	106	107	104	106	104	104	101	104	104	102
22	103	104	104	103	103	104	-	-	-	-	102	102	103	103	104	104	103	-	-	-	-	-	-

Sound pressure level in decibels referenced to 0.0002 dynes per square centimeter

* Maximum sound pressure level reduction due to lowering engine speed (93 to 84 decibels).

TABLE 6

FLY-OVER TESTS - CONDITIONS AND RESULTS

		Configuration I	Configuration II	Configuration III
Speed	knots	60	60	60
Approx. altitude	feet	50	50	50
Rotor tip speed	f.p.s.	720	724	724
Gross weight	lbs	6400	6200	6400
Wind direction	-	S	SSW	WSW
Velocity	knots	14	22 (to 32)	8
Temperature	°F	84.2	80.7	79.3
Relative humidity	%	52	64	66
Over-all Sound Pressure Level During Approach:				
Distance from Microphone:				
	300 feet	91 db	92 db	85 db
	200 feet	95	96	87
	100 feet	98	100	92
	0 feet	107	108	103

TABLE 7

ESTIMATED LOUDNESS LEVELS FOR VARIOUS HU-1A TAIL ROTOR MODIFICATIONS

Design V_{stall} = 150 Kn, T_{hover} = 313 lbs, Distance = 200 ft

	Standard	Mod a	Mod b	Mod c	Mod d
Number of blades	2	3	4	2	4
Diameter - feet	8.4	8.4	8.4	8.4	8.4
Chord - inches	8.4	8.4	8.4	21	10.5
V_t- f.p.s.	710	635	600	590	590
Engine r.p.m.	6400	5730/6400*	5410/6400*	5320/6400*	5320/6400*
Rotational Loudness Level, phons**	81	74	71	74	70
Vortex Loudness Level, phons	68	63	61	64	60
New blades required	-	no/no	no/no	yes/yes	yes/yes
New tail rotor gear box required	-	no/yes	no/yes	no/yes	no/yes
New hub required	-	yes/yes	yes/yes	yes/yes	yes/yes
Relative weight	1	1-1/2	2	2+	2+

* Restricted operation - for test purposes only

** Values shown include a 10 phon increase over calculated values to account for discrepancies between theory and test data (See Section V-D-1-c)

TABLE 8

HU-1A AND HU-1B PERFORMANCE AND HOVERING LOUDNESS LEVELS
WITH VARIOUS MAIN ROTOR CONFIGURATIONS

CONFIGURATION	UNITS	STANDARD HU-1A	MOD 1	MOD 2	MOD 3	STANDARD HU-1B	MOD 4	MOD 5	MOD 6
			←	TO HU-1A	→		←	TO HU-1B	→
Main Rotor									
Number of Blades		2	2	3	3	2	3	2	2
Diameter	ft	43.75	44	44	44	44	44	44	44
Chord	in	15.2	21	21	15.2	21	21	27	31.5
Weights									
Empty	lb	3940	3954	4251	4246	4465	4762	4552	4615
Full Fuel	lb	813	813	813	813	1072	1072	1072	1072
Crew & Misc.	lb	657	657	657	657	657	657	657	657
Payload	lb	1000	1000	1000	1000	1000	1000	1000	1000
Gross	lb	6410	6424	6721	6716	7194	7491	7281	7344
Blade Loading	lbs/sq.ft.	116	83	58	77	93	64	74	64
Tip Speed	f.p.s.	652*/720	652/720	652/720	652/720	656/724	656	656	656
Hovering Peak Loudness Level									
Vortex	phons	74/79	69/76	69/73	70/75	72/79	70	71	68
Rotational	phons	63/67	64/70	66/72	63/69	67/77	67	71	73
Hov. Ceiling-Std Day	ft	7000/8000	8200/7800	6500/5400	6300/5700	9000/9500	8200	8500	8900
Max R/C-S.L. Std Day	f.p.m.	1720/1800	1880/1800	1500/1240	1640/1570	1900/2000	1800	1840	1860
Range with Payload	n.mi.	124/146	151/150	146/138	147/145	207/210	199	202	200
Max. Stall Limit Vel.	knots	76/117	108/130	122/136	12[illegible]/155	93/129	119	115	120
New Blades Required	-	-	no	no	no	-	no	yes	yes
New Hub Required	-	-	no	yes	yes	-	yes	no	yes

*Low tip speed for HU-1A recommended only at low forward speeds (< 70 Kn.)

TABLE 9

EFFECTS OF DIAMETER ON MAIN ROTOR LOUDNESS LEVELS

Configuration			Diameter Feet	Tip Speed F.P.S.	Blade Loading Lbs/Sq Ft	Maximum Loudness Level Phons
HU-1B						
Number of blades		2	48	717	85	77
Chord	-in	21	44 (std)	656	93	72
Gross weight	-lb	7152	40	595	102	68
Engine speed	-r.p.m.	5800				
Modification 4						
Number of blades		3	44	656	64	69
Chord	-in	21	38	567	74	62
Gross weight	-lb	7449				
Engine speed	-r.p.m.	5800				
Modification 6						
Number of blades		2	44	656	63	73
Chord	-in	31.5	38	567	73	63
Gross weight	-lb	7202				
Engine speed	-r.p.m.	5800				

TABLE 10

SUMMARY OF OPTIMUM CONFIGURATIONS

CONFIGURATION	BASED ON HU-1A		BASED ON HU-1B		
	BASIC HU-1A	MOD 1-MAIN ROTOR MOD b-TAIL ROTOR	BASIC HU-1B	MOD 5-MAIN ROTOR MOD b-TAIL ROTOR	MOD 4-MAIN ROTOR MOD b-TAIL ROTOR
Number of main rotor blades/chord, in.	2/15.2	2/21	2/21	2/27	3/21
Number of tail rotor blades/chord, in.	2/8.4	4/8.4	2/8.4	4/8.4	4/8.4
Engine speed, r.p.m.	6400	5800	6400	5800	5800
Mission gross weight, lb.	6410	6424	7194	7281	7491
Relative Performance (in percent)	(REF)				
Hovering ceiling	100	102.5	118.8	106.3	102.5
Range	100	103.4	143.8	138.3	136.3
Relative Loudness (in percent)					
Most prominent component (tail rotor rotational noise)	100	50	100	50	50
Total	100	56	117	66	60

ILLUSTRATIONS

FIGURE 1. HU-1A HELICOPTER.

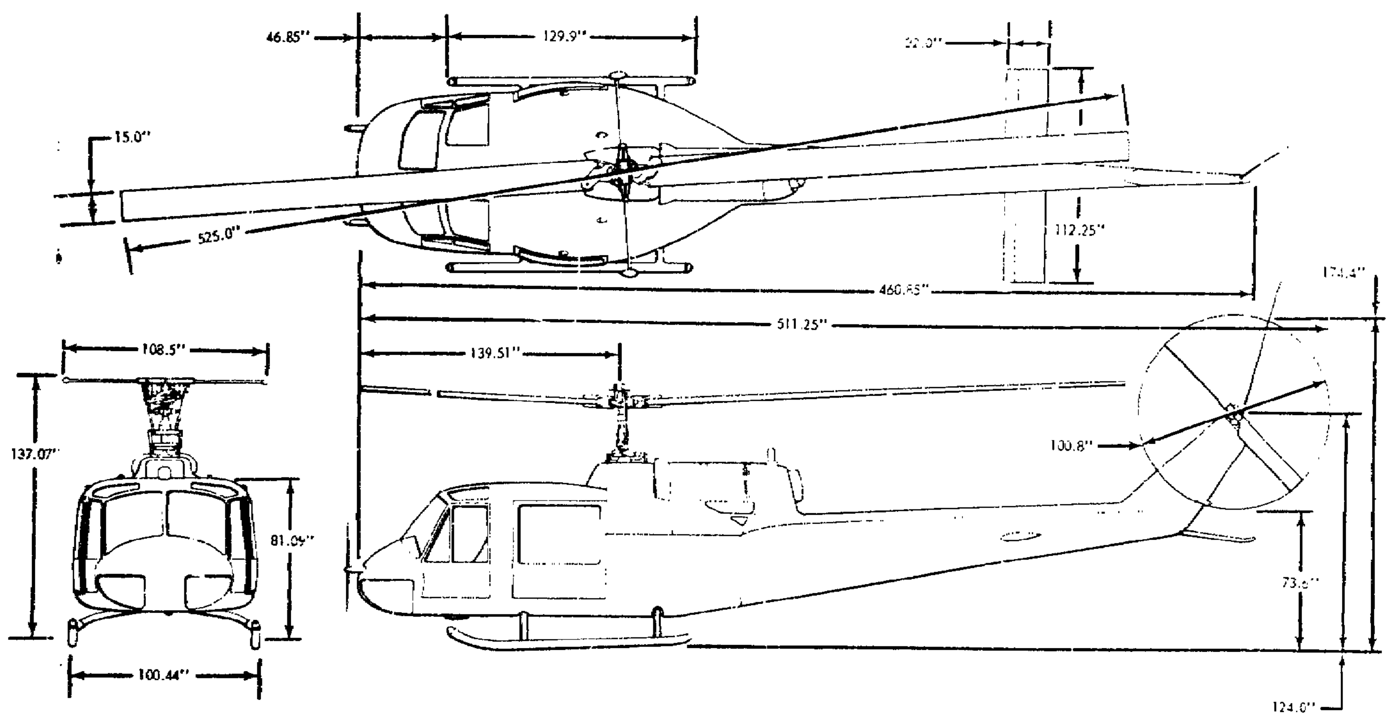


FIGURE 2. HU-1A HELICOPTER – THREE-VIEW DIMENSIONAL DATA.

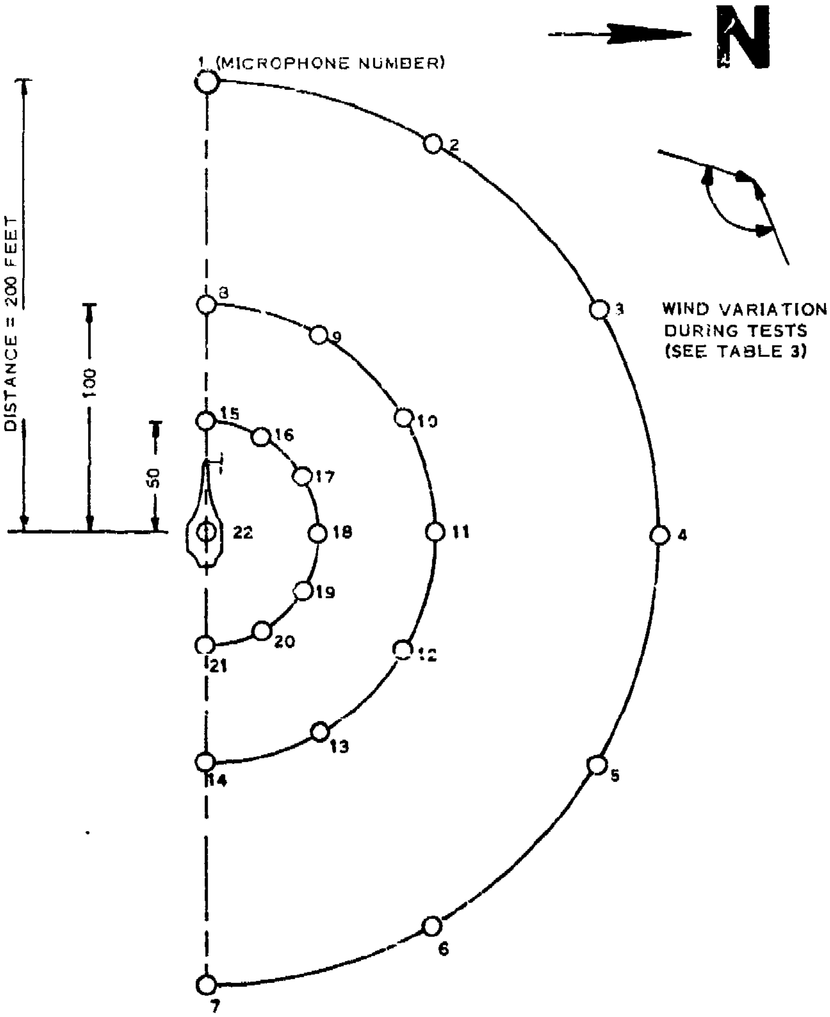


FIGURE 3. TIEDOWN AND HOVER MICROPHONE LOCATIONS.

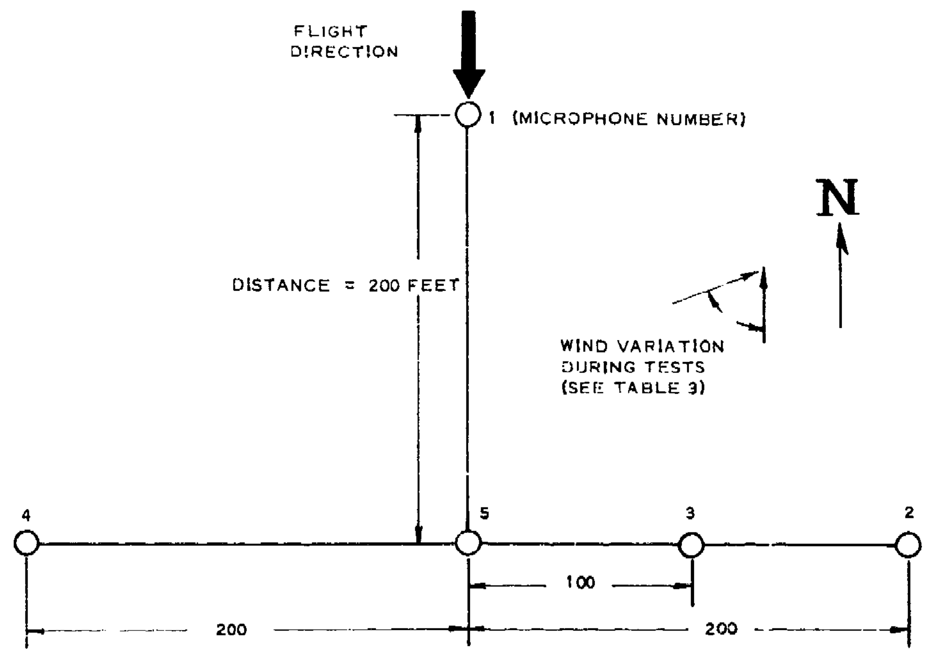


FIGURE 4. FLY-OVER MICROPHONE LOCATIONS.

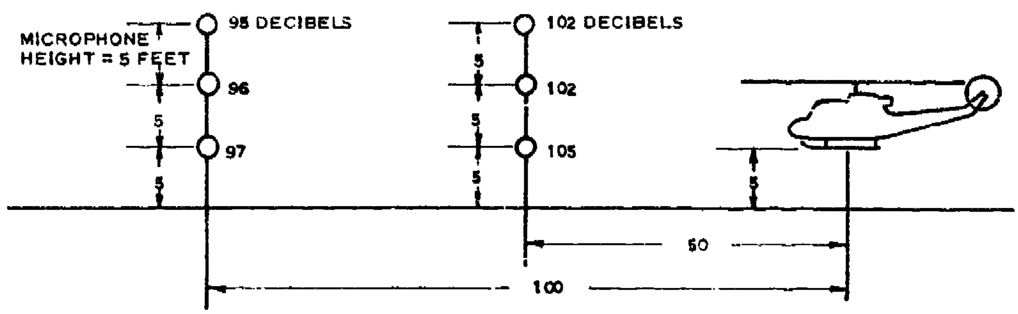


FIGURE 5. EFFECT OF MICROPHONE HEIGHT ON OVER-ALL SOUND PRESSURE LEVEL.

FIGURE 6. ACOUSTICAL DATA RECORDING SYSTEM
(GENERAL DYNAMICS/FORT WORTH).

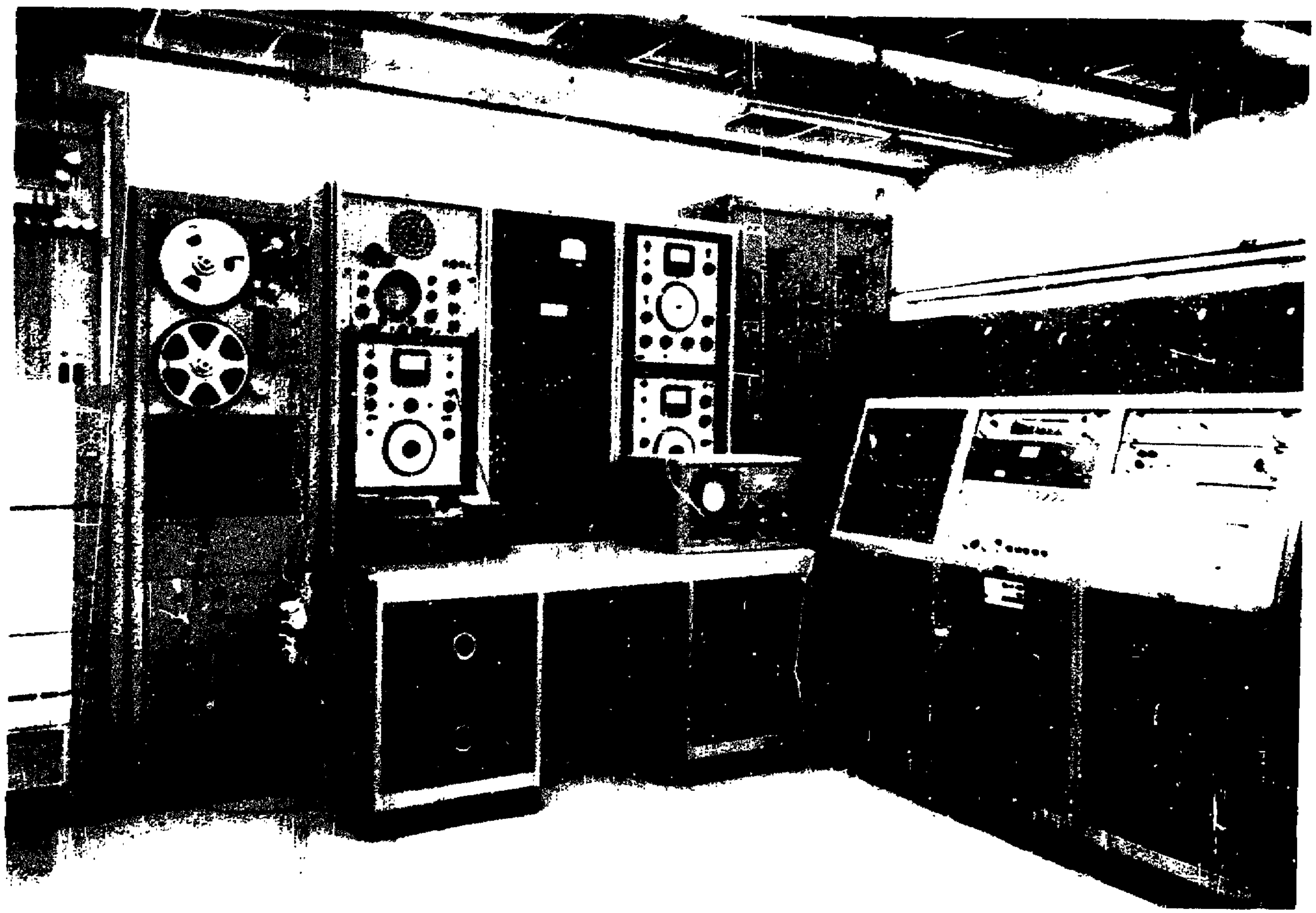


FIGURE 7. ACOUSTICAL DATA REDUCTION SYSTEM
(GENERAL DYNAMICS/FORT WORTH).

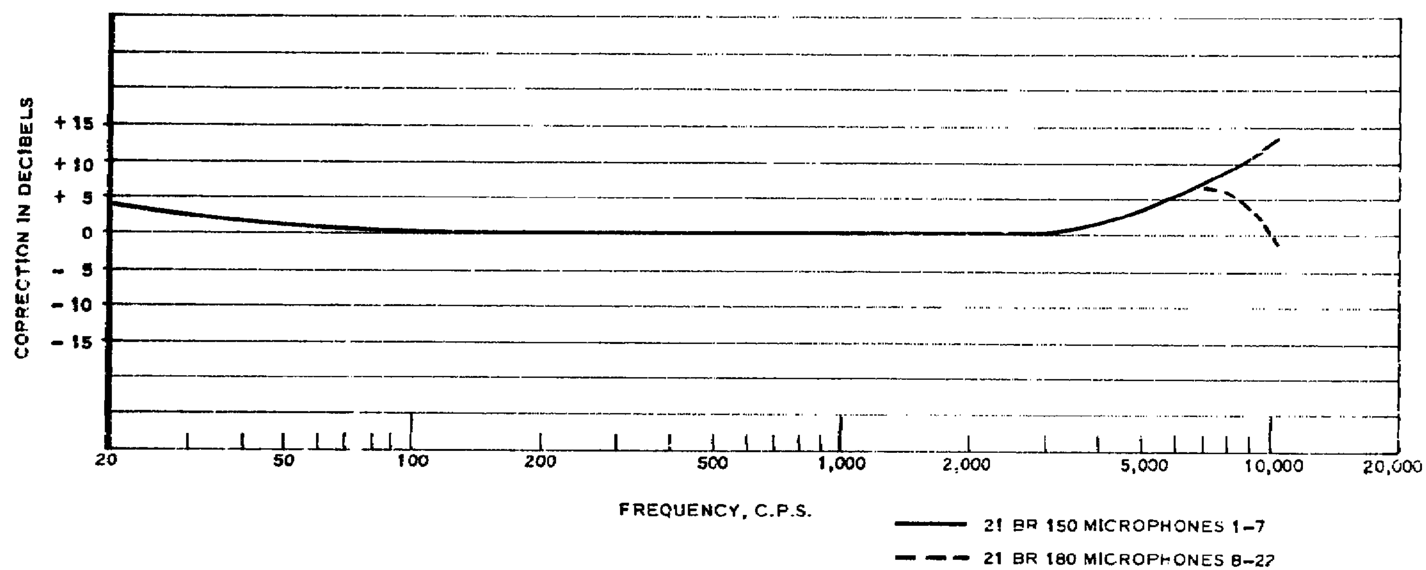


FIGURE 8. CORRECTIONS TO BE APPLIED TO ALL DATA REDUCED
BY 6-C.P.S. CONSTANT BANDWIDTH ANALYSIS.

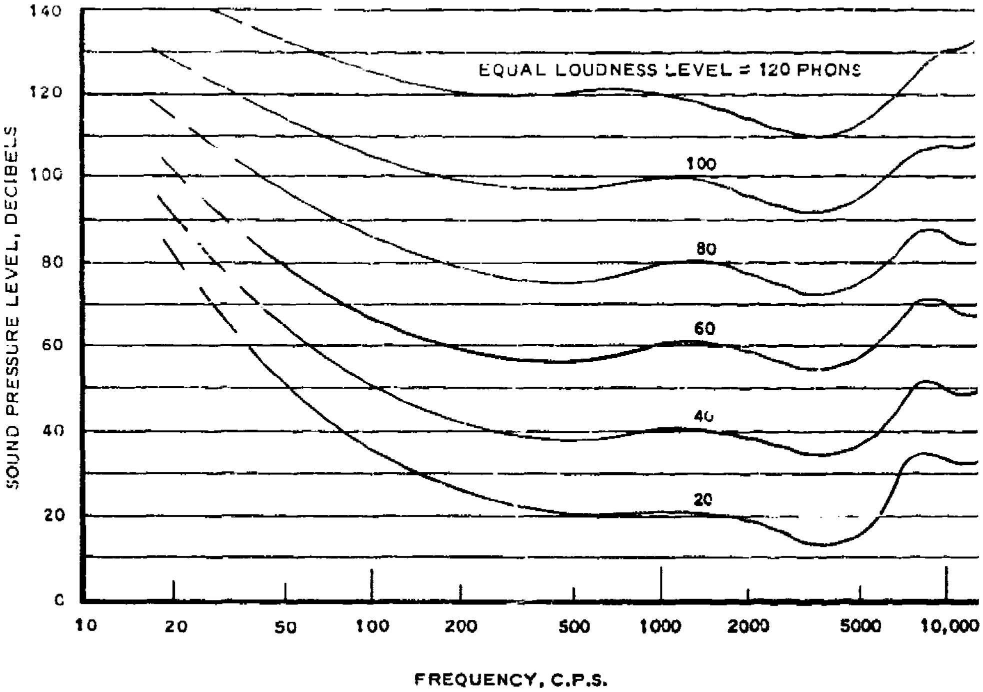


FIGURE 9. EQUAL LOUDNESS LEVEL CONTOURS (REFERENCE 9).

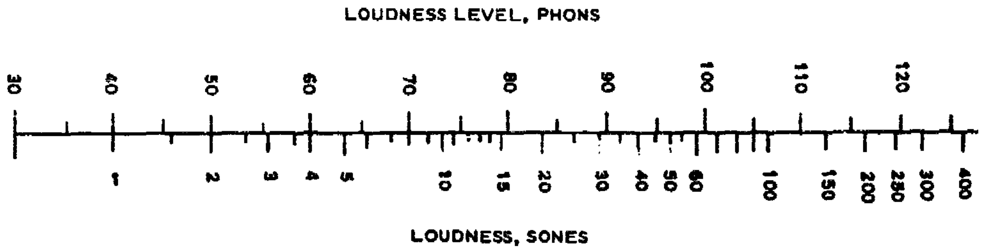


FIGURE 10. LOUDNESS LEVEL AND LOUDNESS NOMOGRAM (REFERENCE 9).

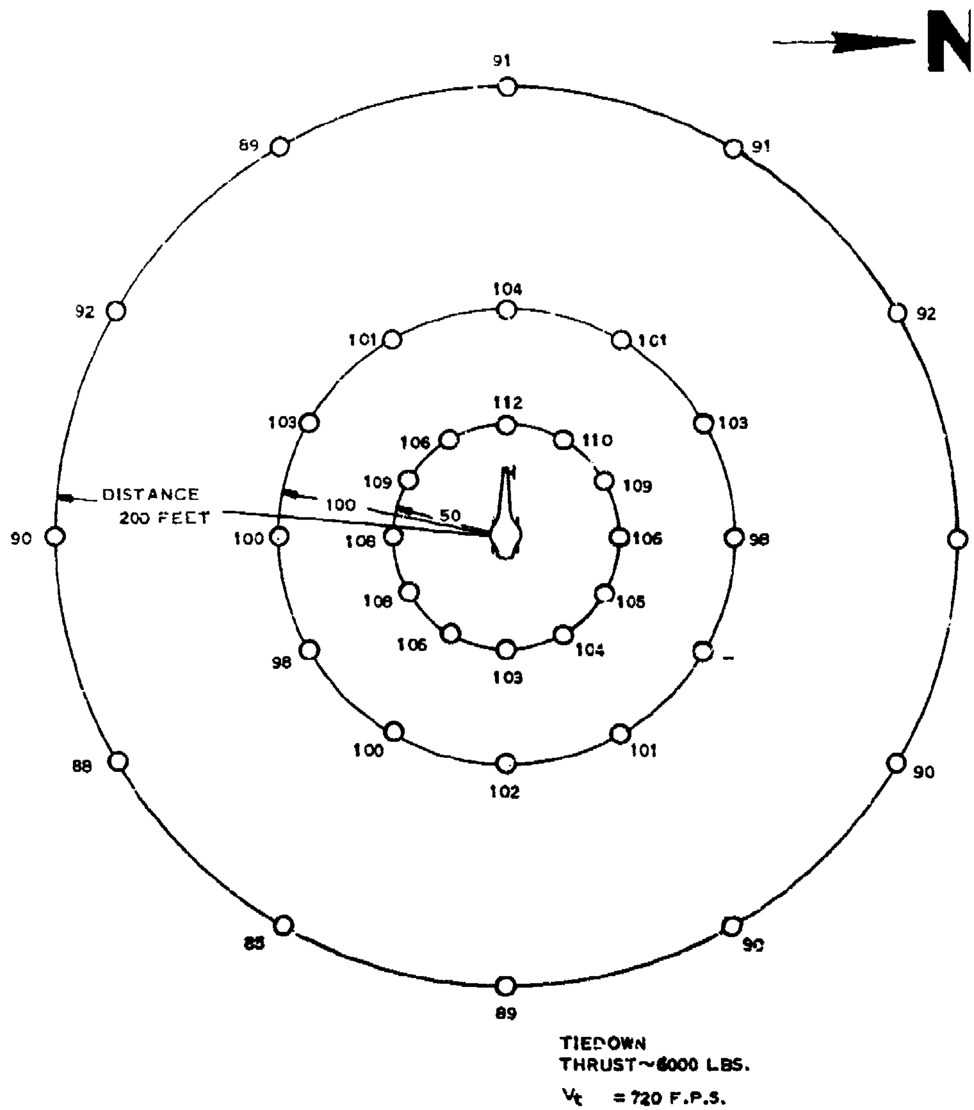


FIGURE 11. HU-1A EXTERNAL OVER-ALL SOUND PRESSURE LEVEL DISTRIBUTION.

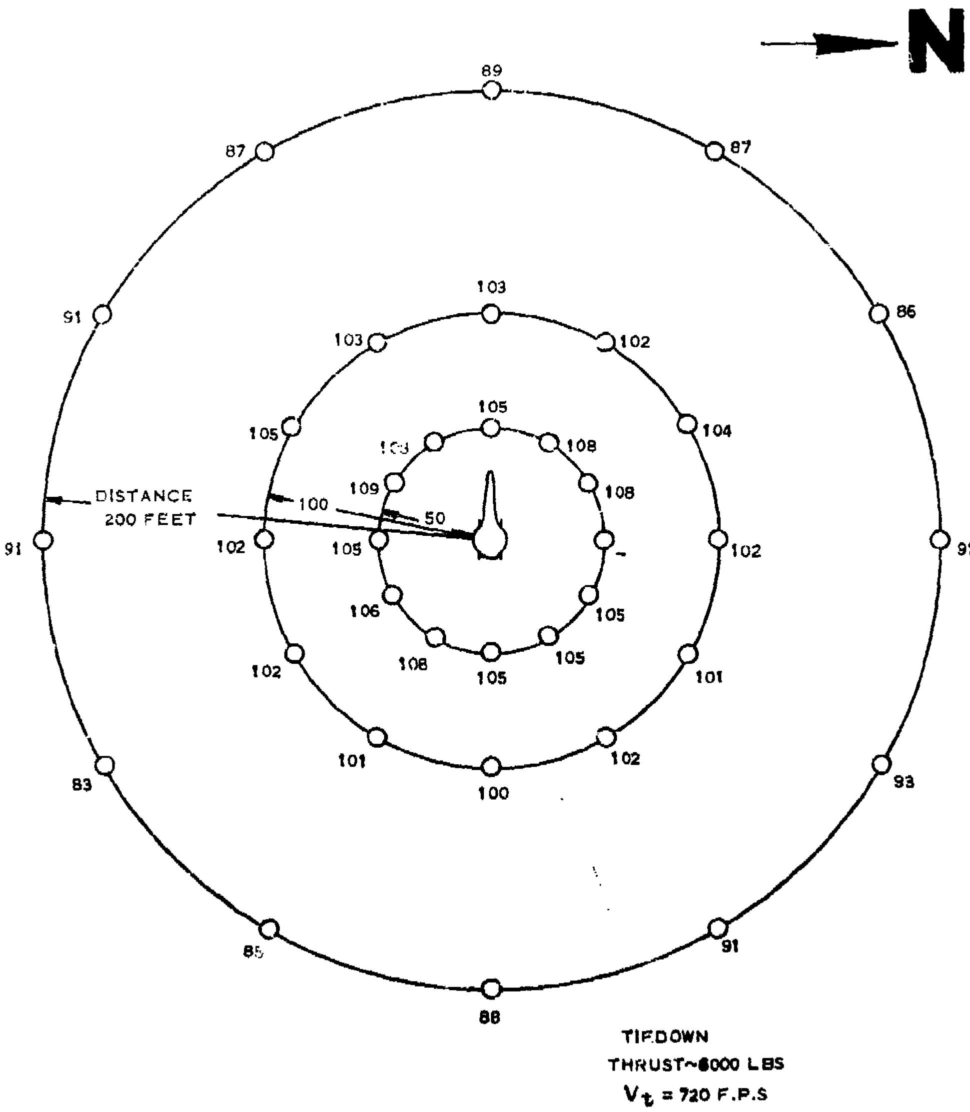


FIGURE 12. HU-1A EXTERNAL OVER-ALL SOUND PRESSURE LEVEL DISTRIBUTION WITH TAIL ROTOR DISCONNECTED.

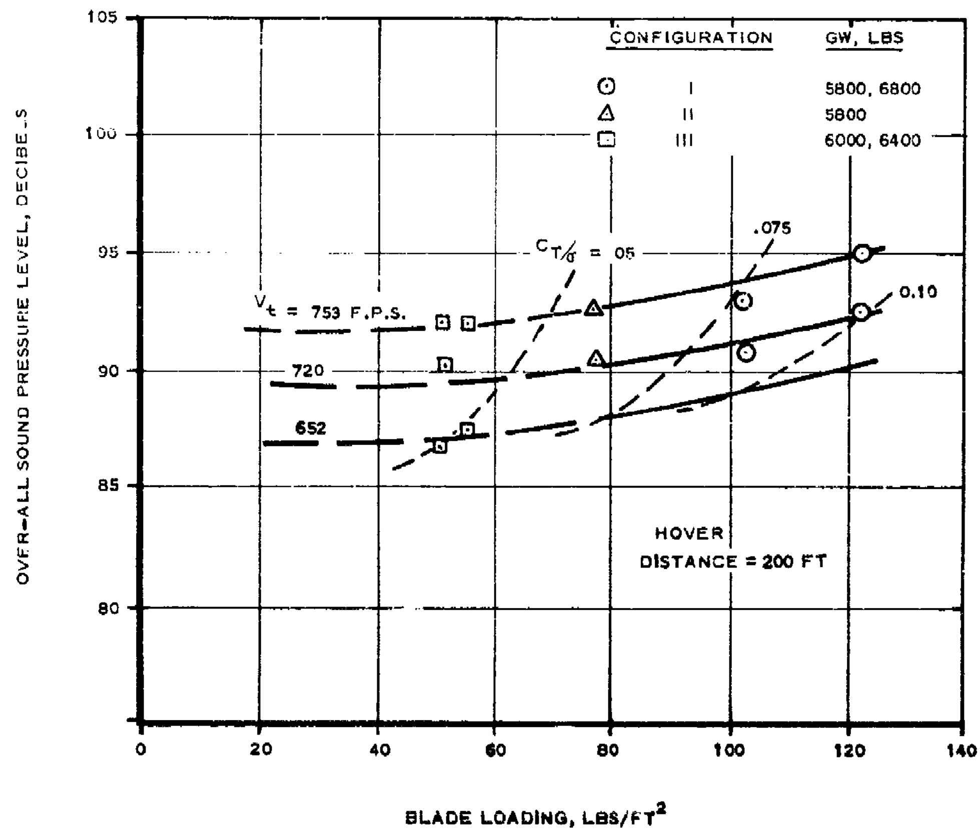


FIGURE 13. MAXIMUM OVER-ALL SOUND PRESSURE LEVEL AS A FUNCTION OF BLADE LOADING AND TIP SPEED.

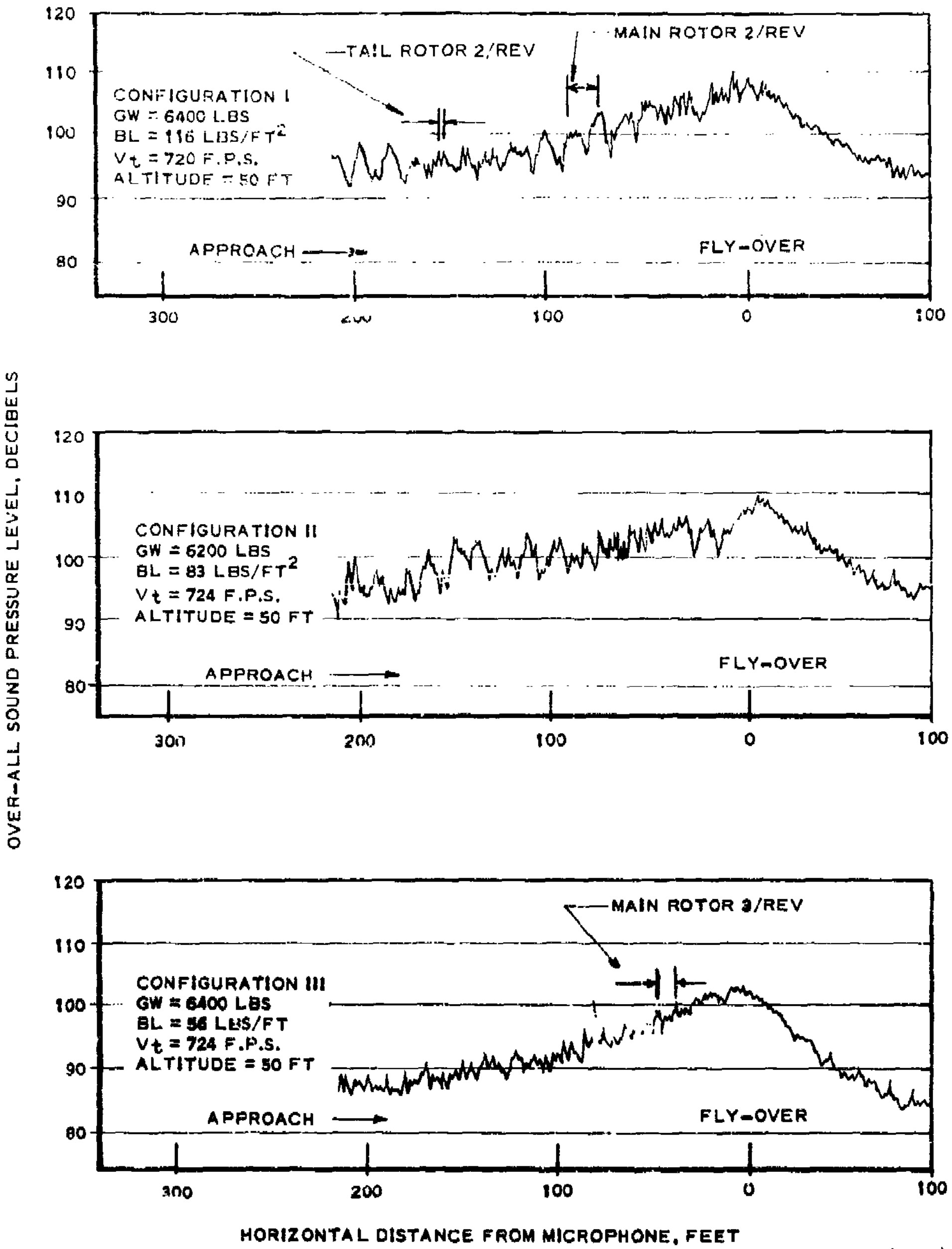
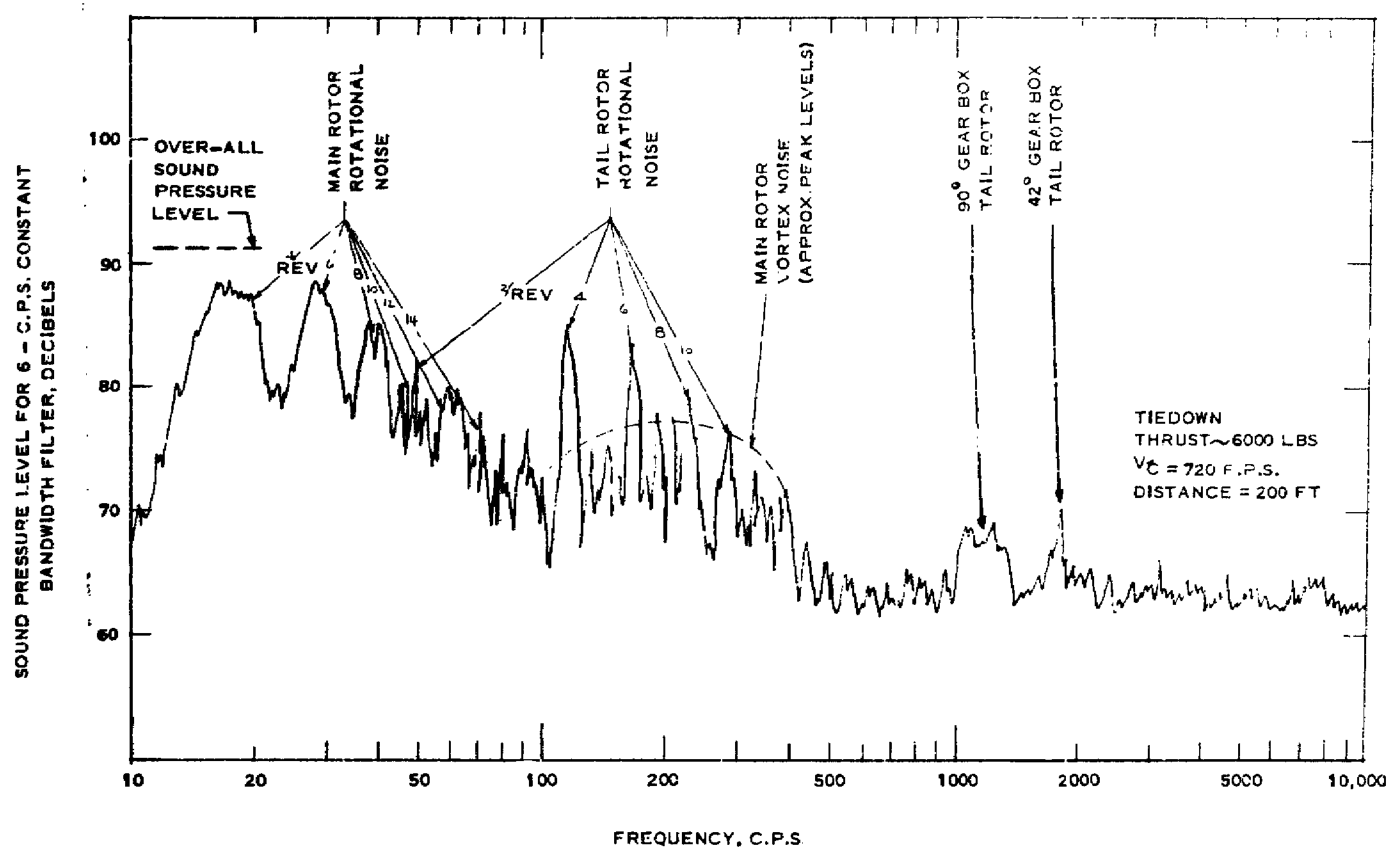


FIGURE 14. GROUND NOISE DURING FLY-OVER (60-KNOT LEVEL FLIGHT).



(UNCORRECTED FOR SYSTEM FREQUENCY RESPONSE, SEE FIGURE 8)

FIGURE 15. HU–1A EXTERNAL NOISE SPECTRUM.

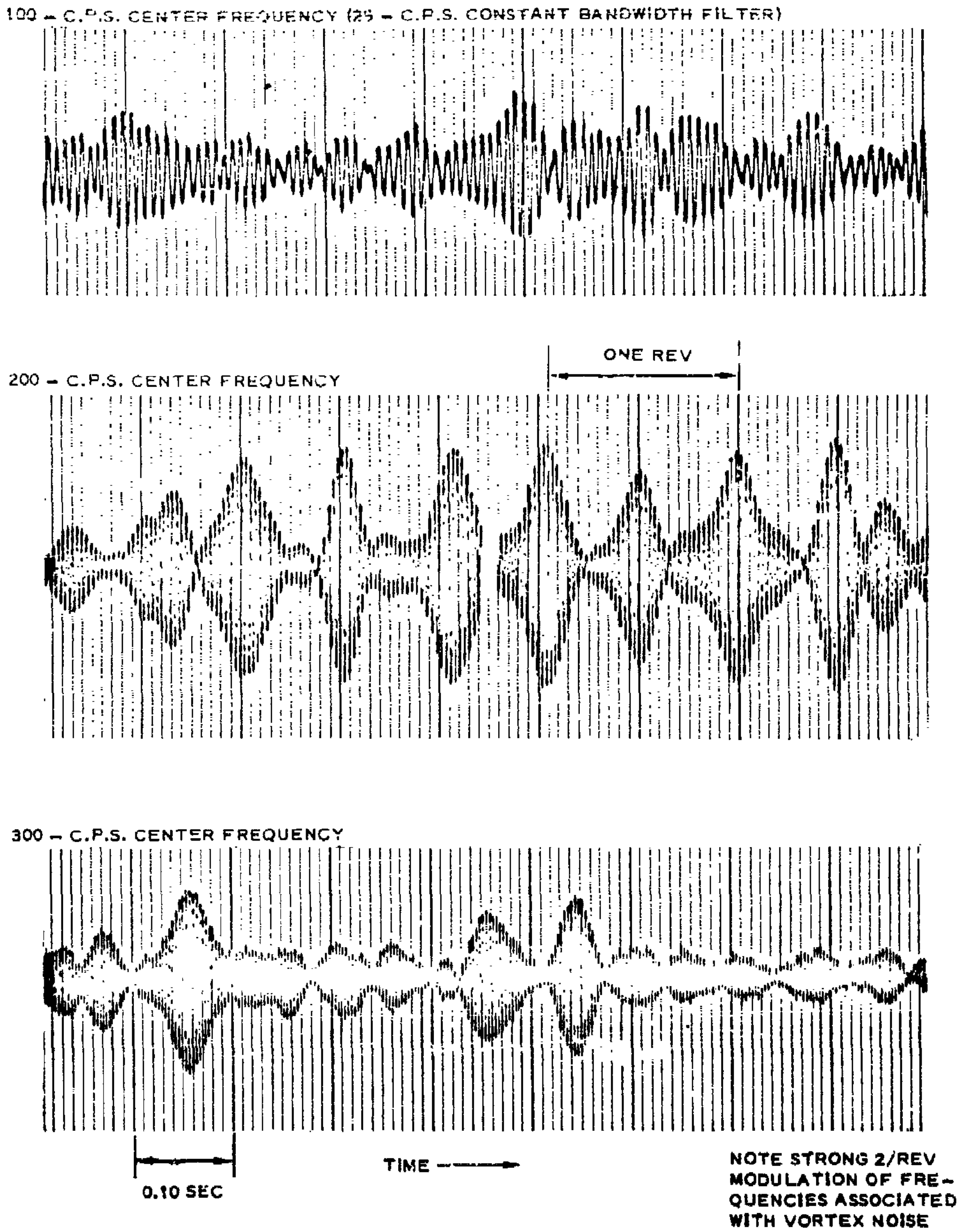


FIGURE 16. TIME HISTORIES OF TWO-BLADED MAIN ROTOR VORTEX NOISE.

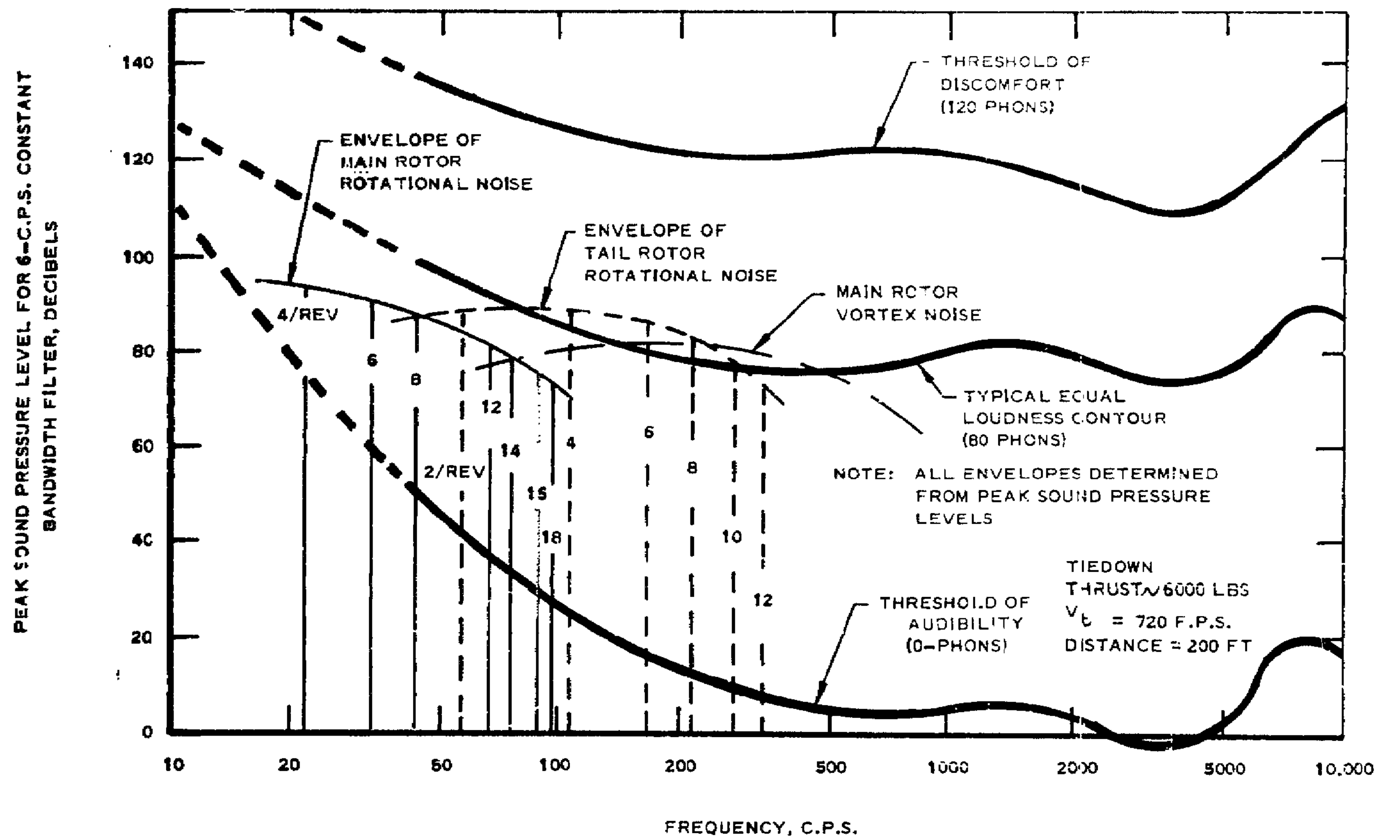


FIGURE 17. RELATIVE LOUDNESS OF MAJOR EXTERNAL NOISE SOURCES OF HU–1A.

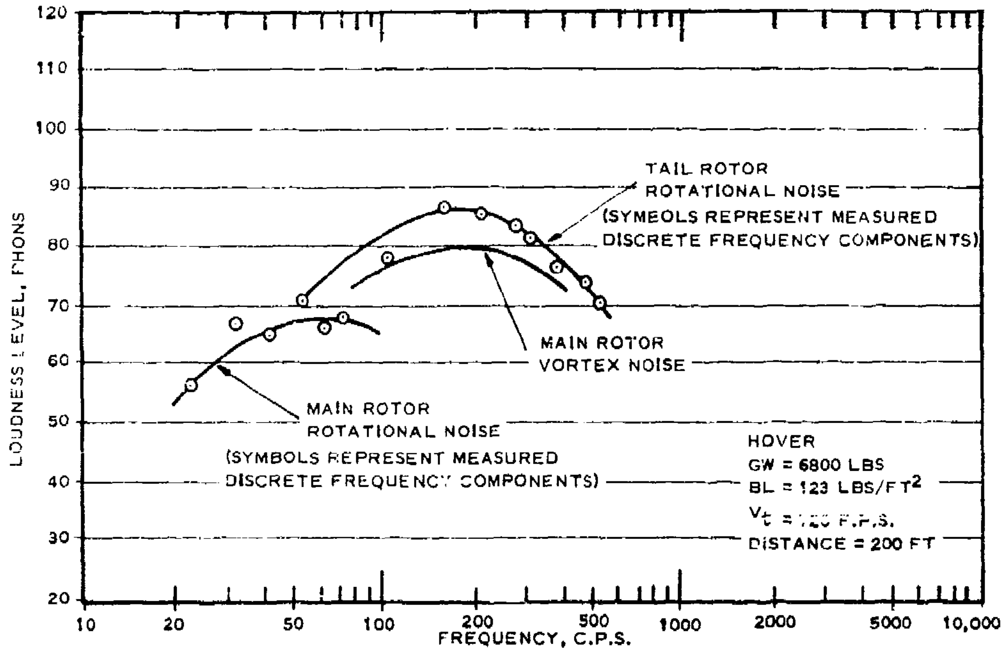


FIGURE 18. LOUDNESS LEVELS OF MAJOR EXTERNAL NOISE SOURCES OF HU-1A.

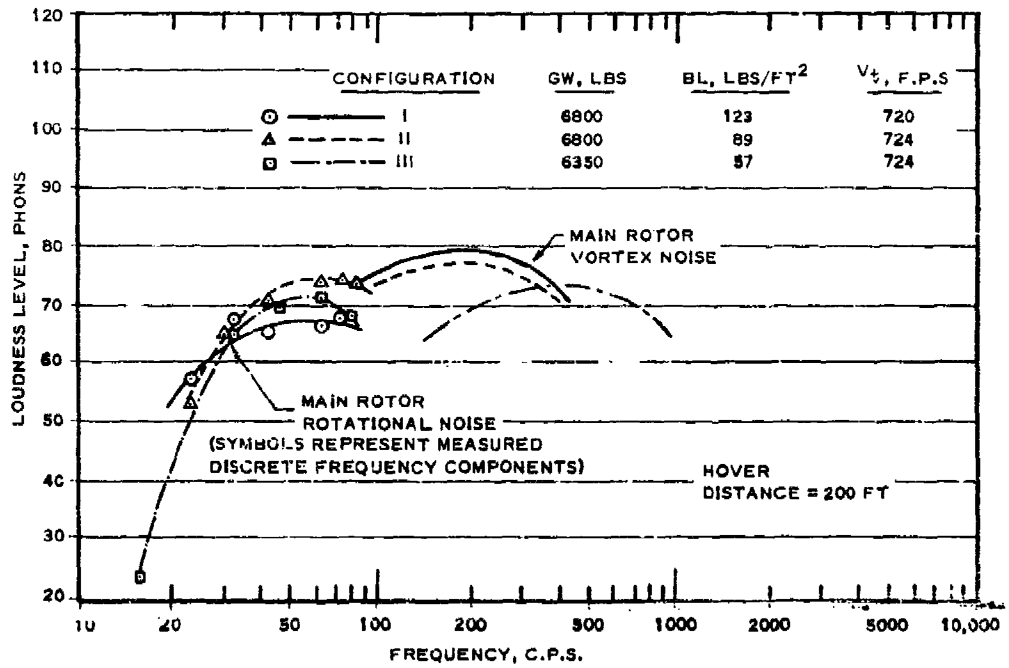


FIGURE 19. EFFECT OF MAIN ROTOR CONFIGURATIONS ON LOUDNESS LEVEL.

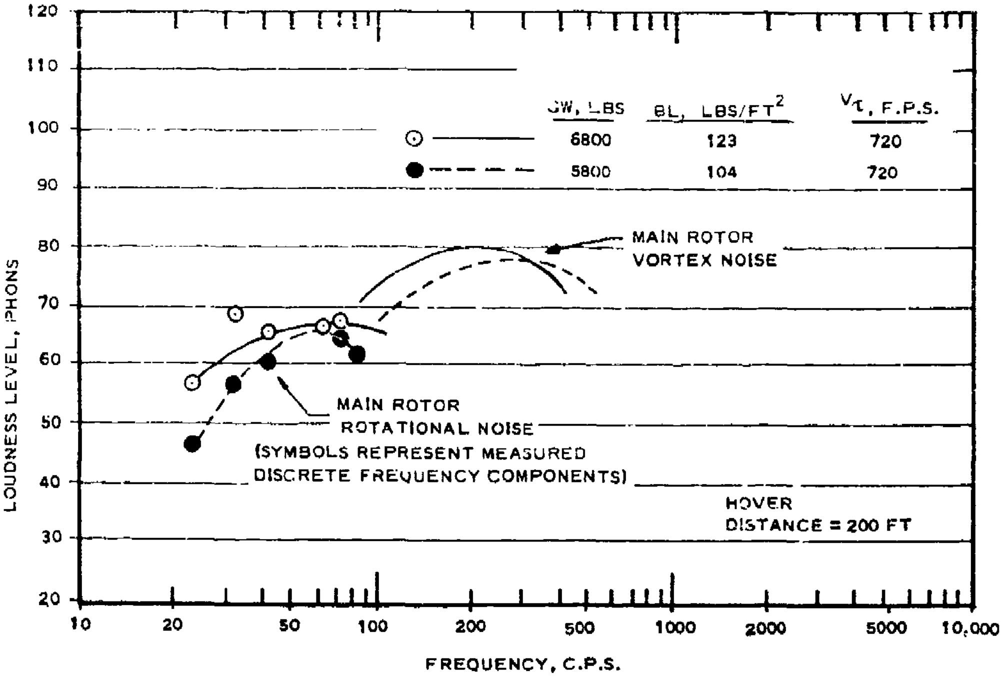


FIGURE 20. EFFECT OF BLADE LOADING AND THRUST ON LOUDNESS LEVEL (CONFIGURATION I).

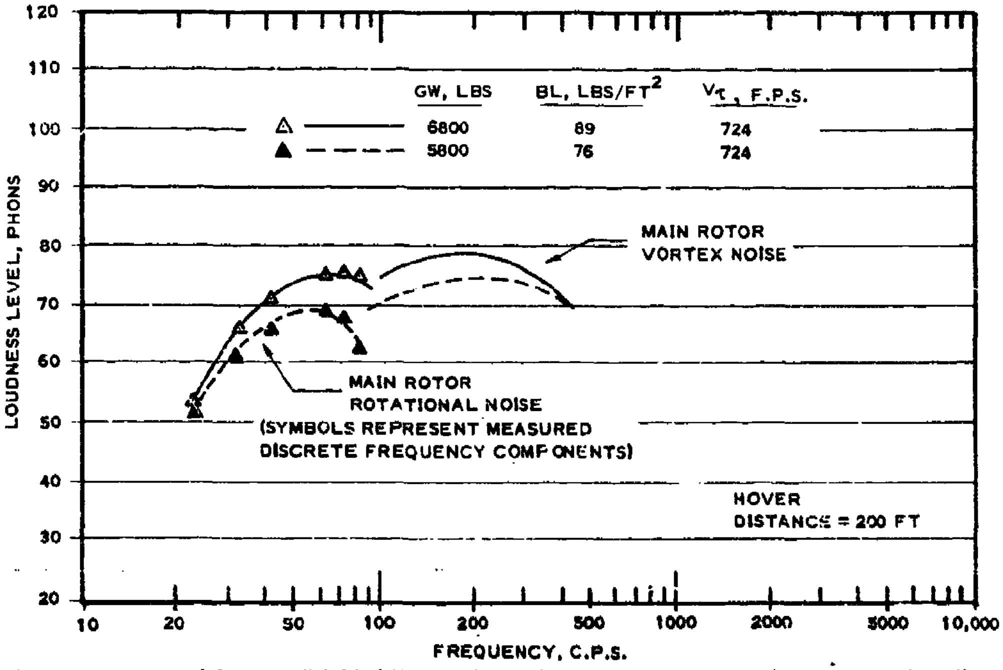


FIGURE 21. EFFECT OF BLADE LOADING AND THRUST ON LOUDNESS LEVEL (CONFIGURATION II).

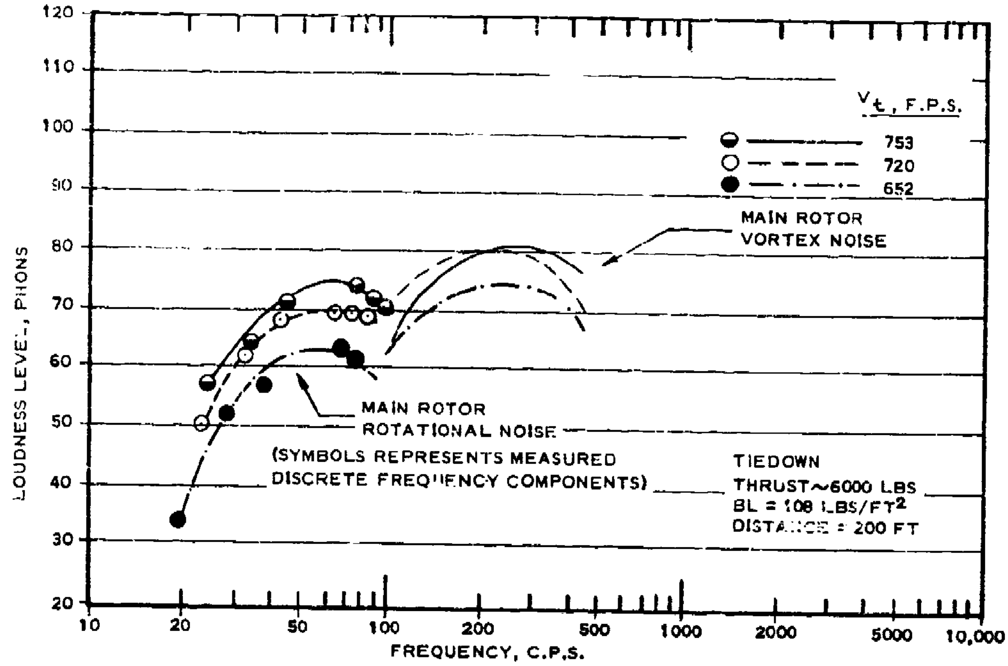


FIGURE 22. EFFECT OF TIP SPEED ON LOUDNESS LEVEL (CONFIGURATION I).

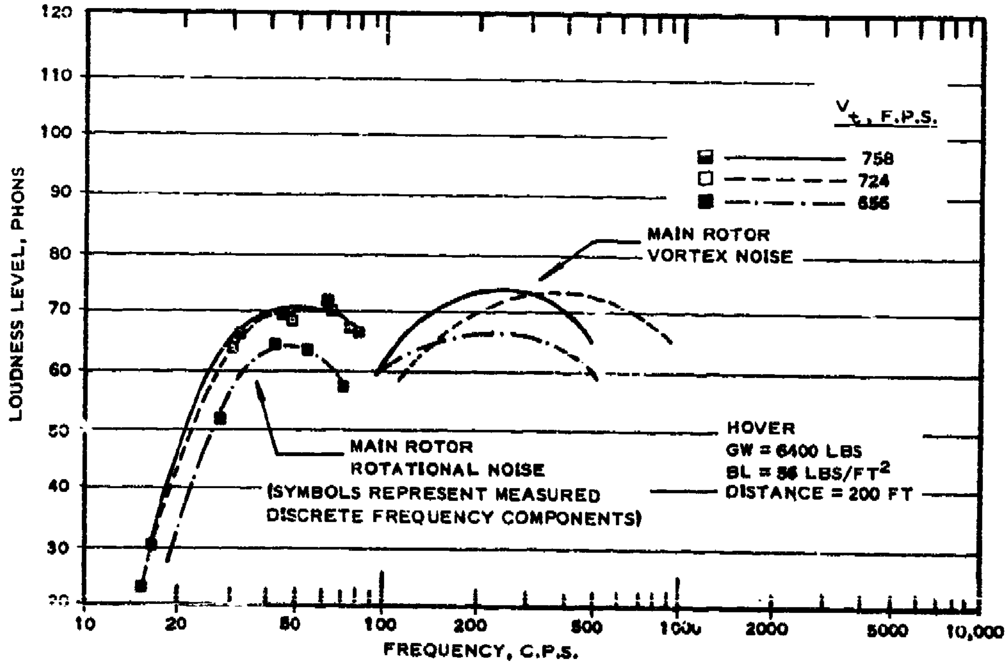


FIGURE 23. EFFECT OF TIP SPEED ON LOUDNESS LEVEL (CONFIGURATION III).

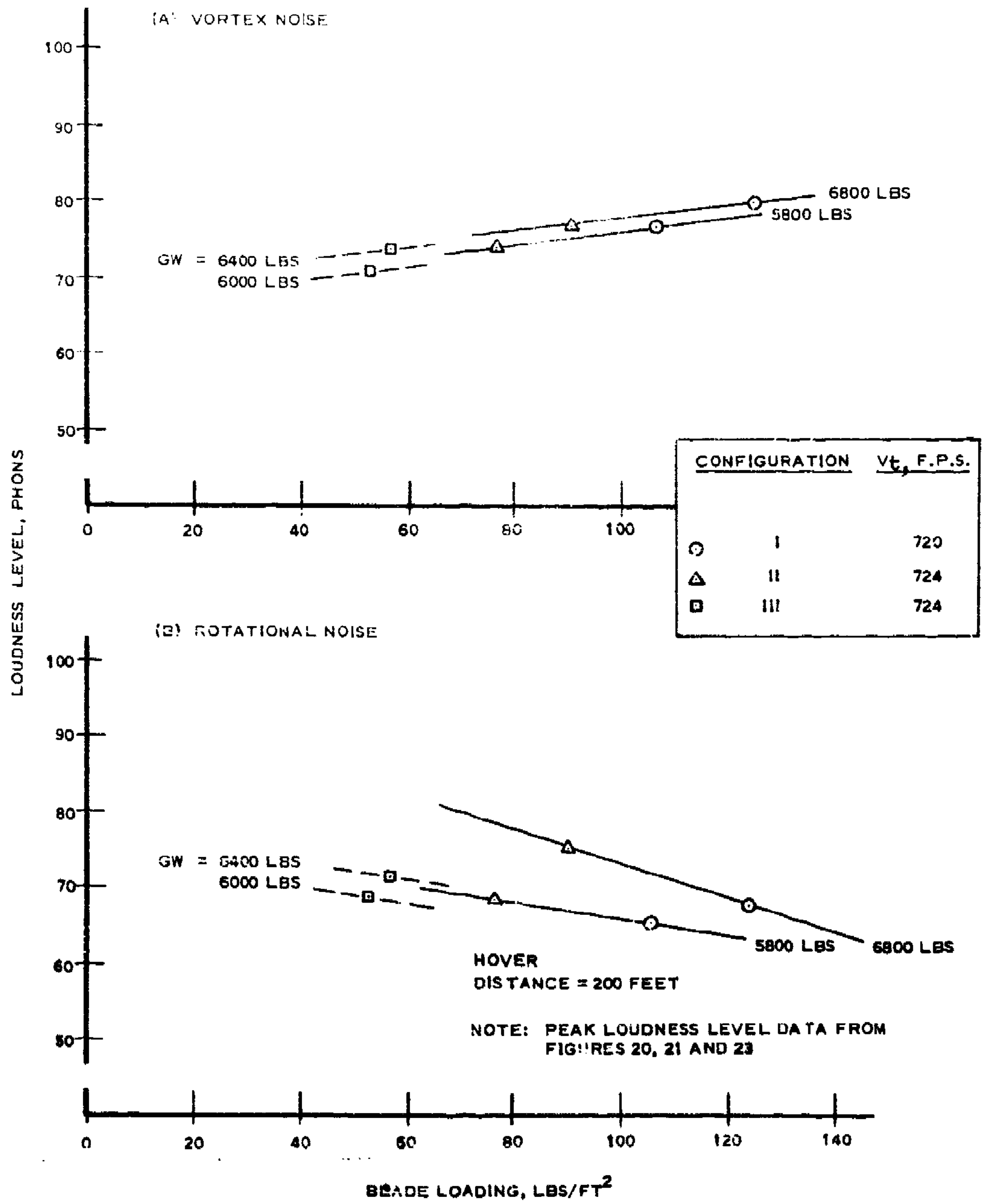


FIGURE 24. EFFECT OF BLADE LOADING AND THRUST ON PEAK LOUDNESS LEVEL.

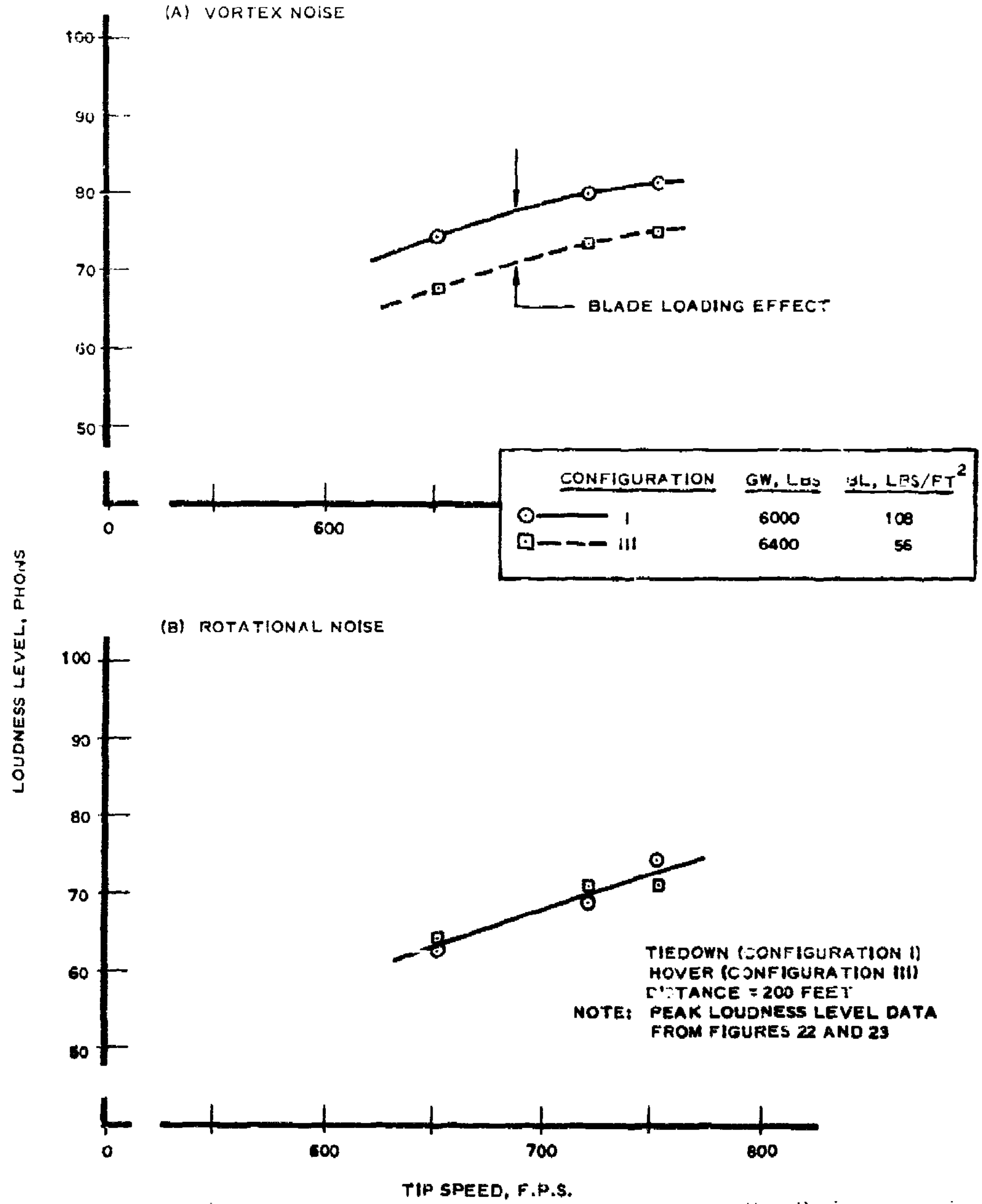


FIGURE 25. EFFECT OF TIP SPEED ON PEAK LOUDNESS LEVEL (CONFIGURATIONS I AND III).

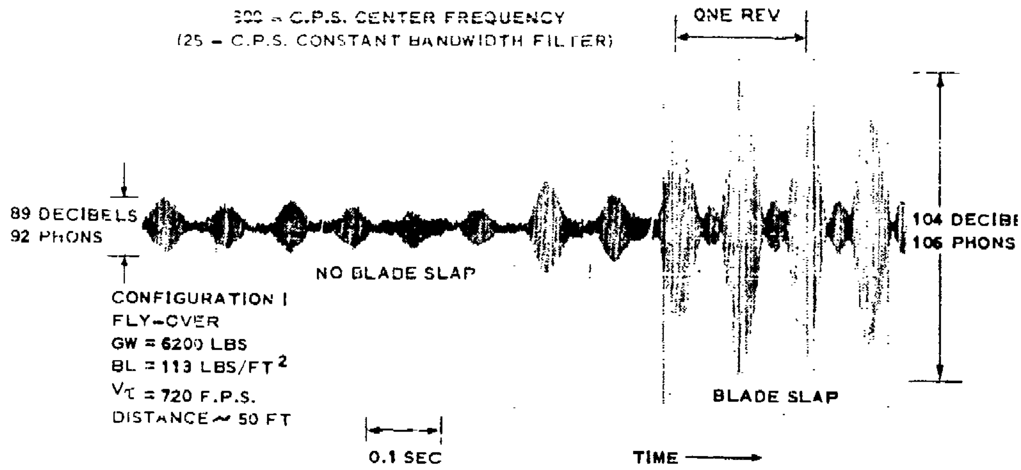


FIGURE 26. SINGLE ROTOR HELICOPTER BLADE SLAP DURING TURN AT A FREQUENCY OF 500 – C.P.S.

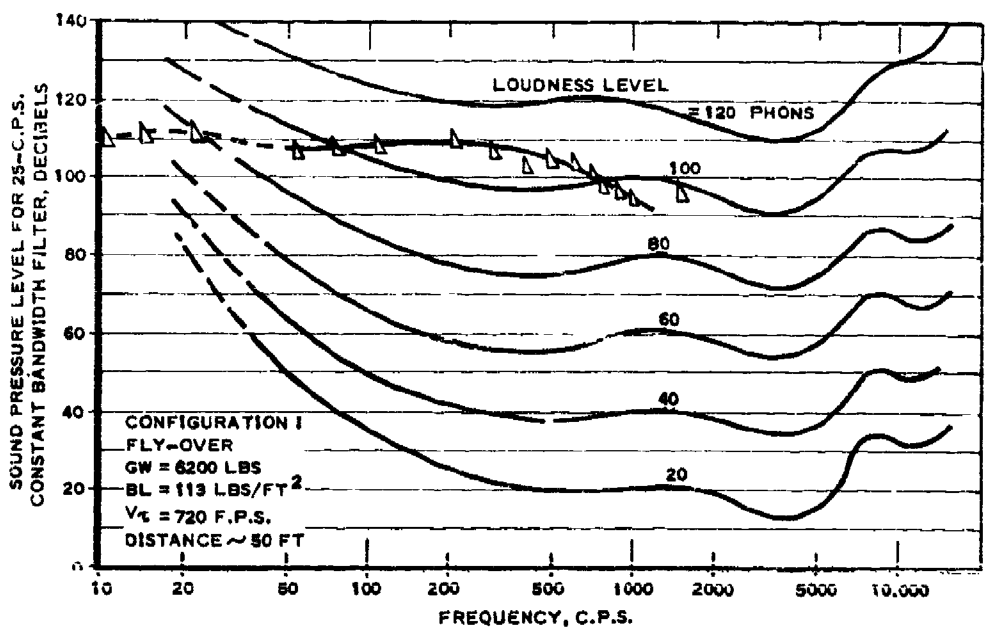


FIGURE 27. LOUDNESS LEVEL OF SINGLE ROTOR HELICOPTER BLADE SLAP

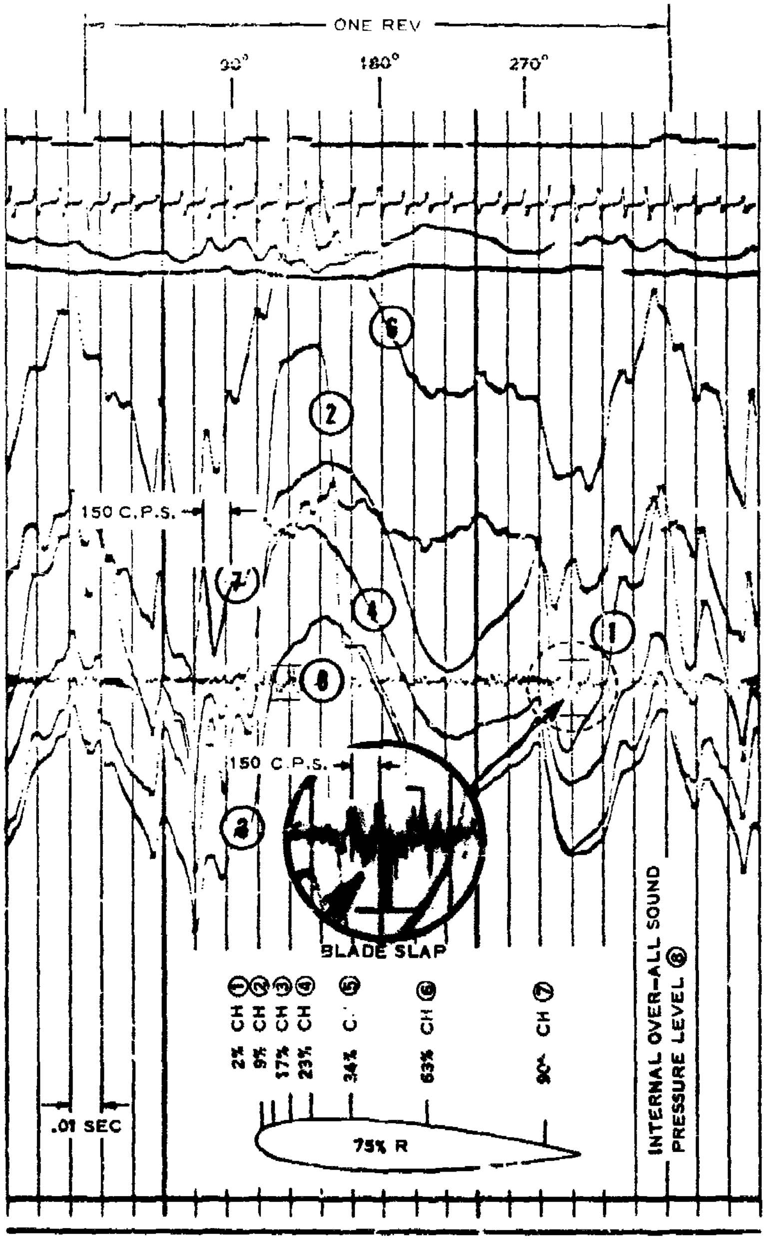


FIGURE 28. BLADE DIFFERENTIAL PRESSURE AND INTERNAL OVER-ALL SOUND PRESSURE LEV DURING BLADE SLAP (75 PER CENT BLADE RADIUS).

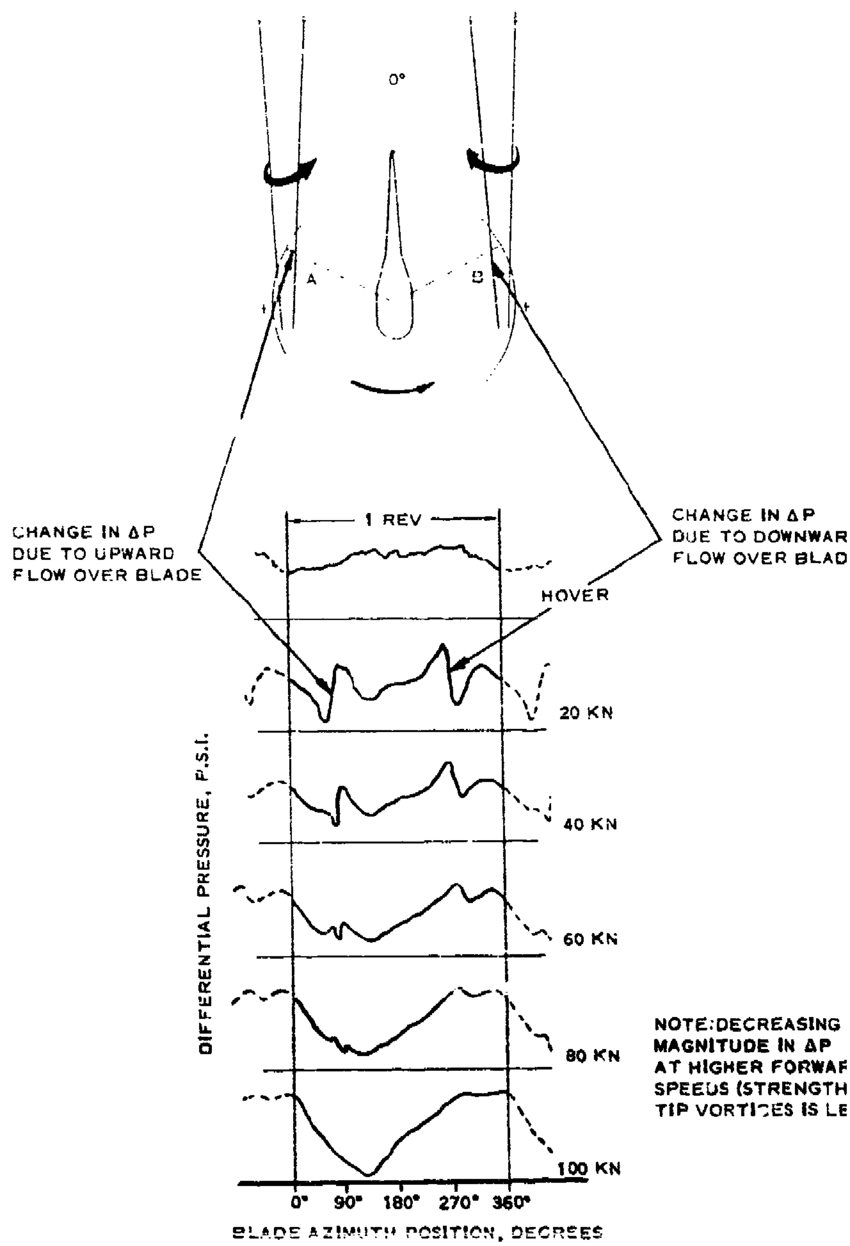


FIGURE 29. DIFFERENTIAL PRESSURE VARIATIONS NEAR THE TIP OF A HELICOPTER MAIN ROTOR.

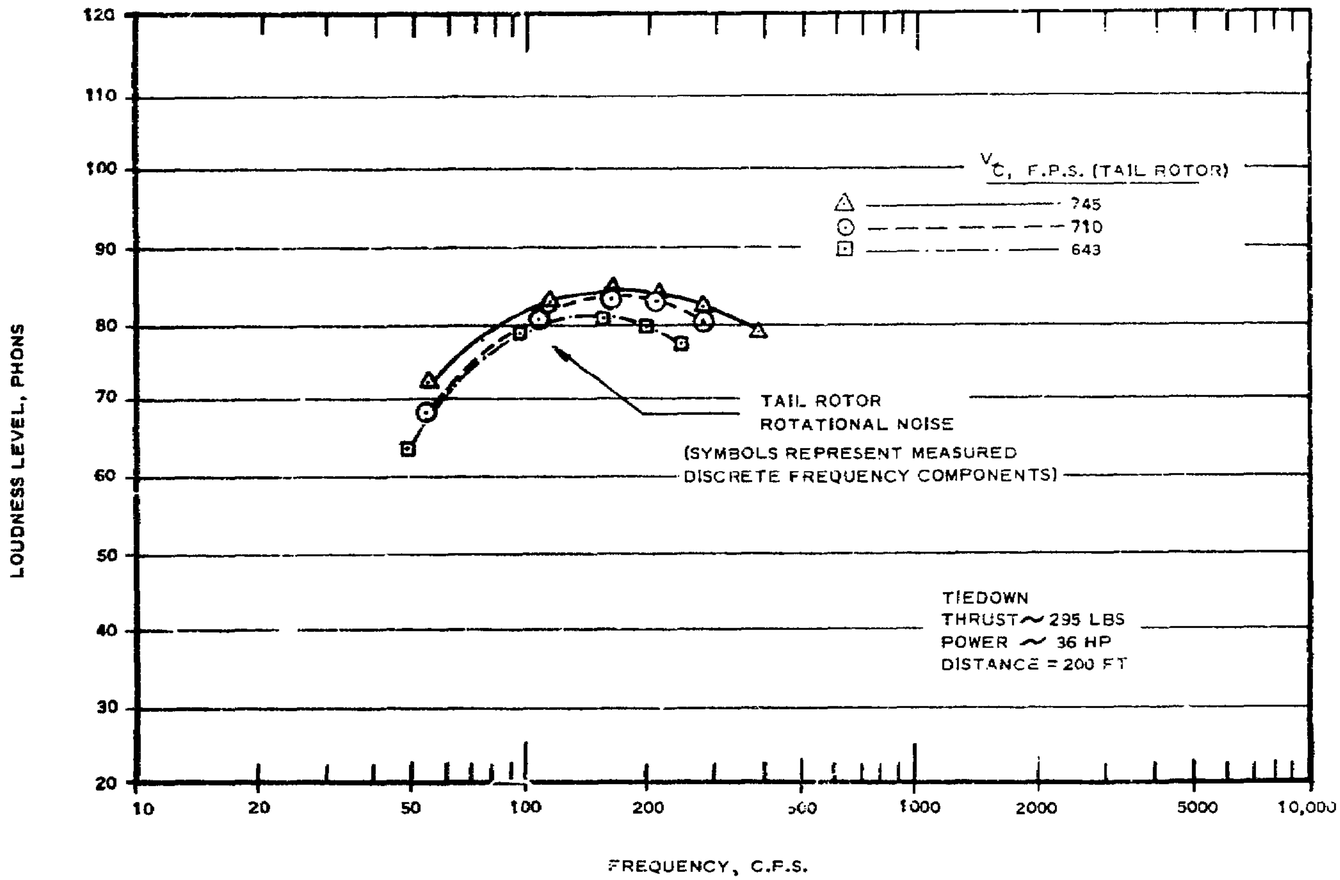


FIGURE 30. EFFECT OF TIP SPEED ON HU-1A TAIL ROTOR LOUDNESS LEVEL.

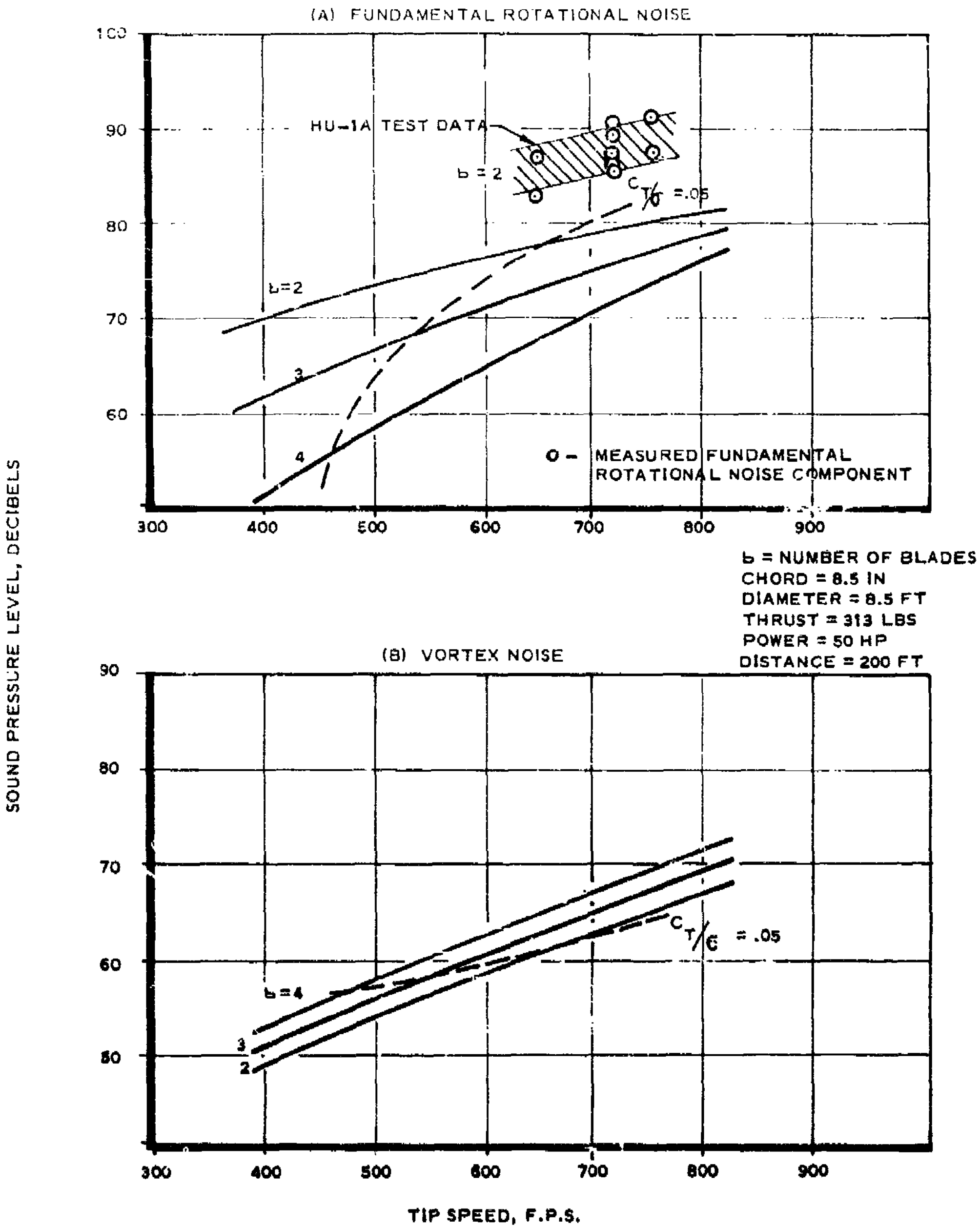


FIGURE 31. CALCULATED TAIL ROTOR ROTATIONAL AND VORTEX NOISE.

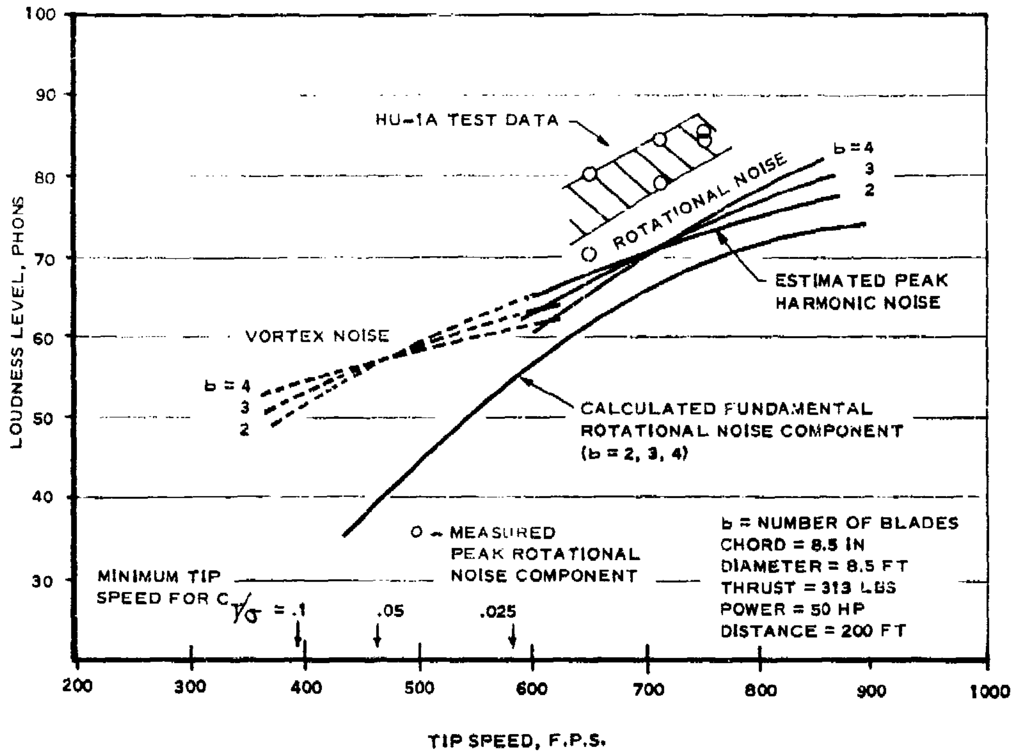


FIGURE 32. ESTIMATED MAXIMUM TAIL ROTOR LOUDNESS LEVEL FOR VARIOUS TIP SPEEDS.

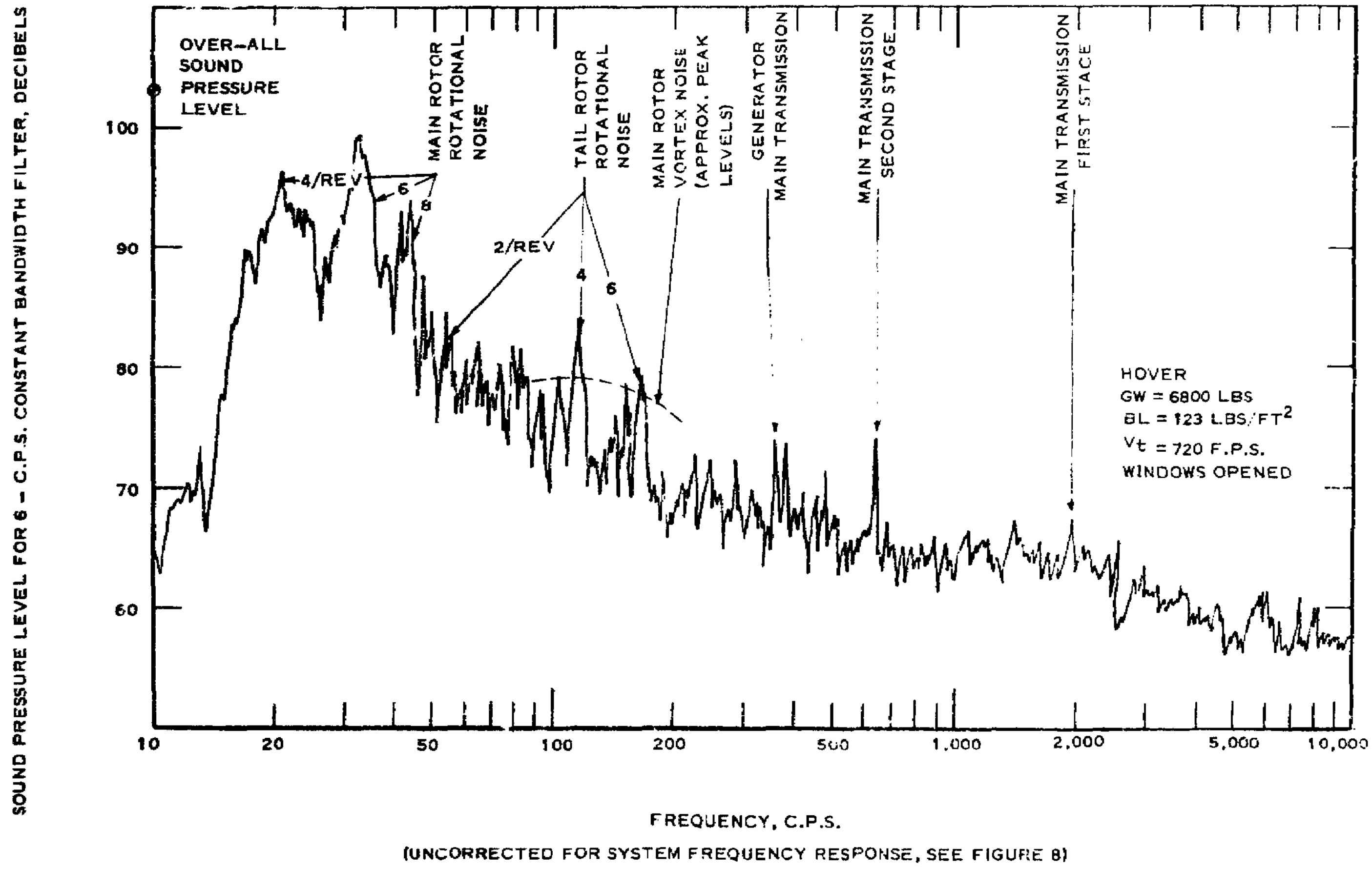


FIGURE 33. HU-1A INTERNAL NOISE SPECTRUM.

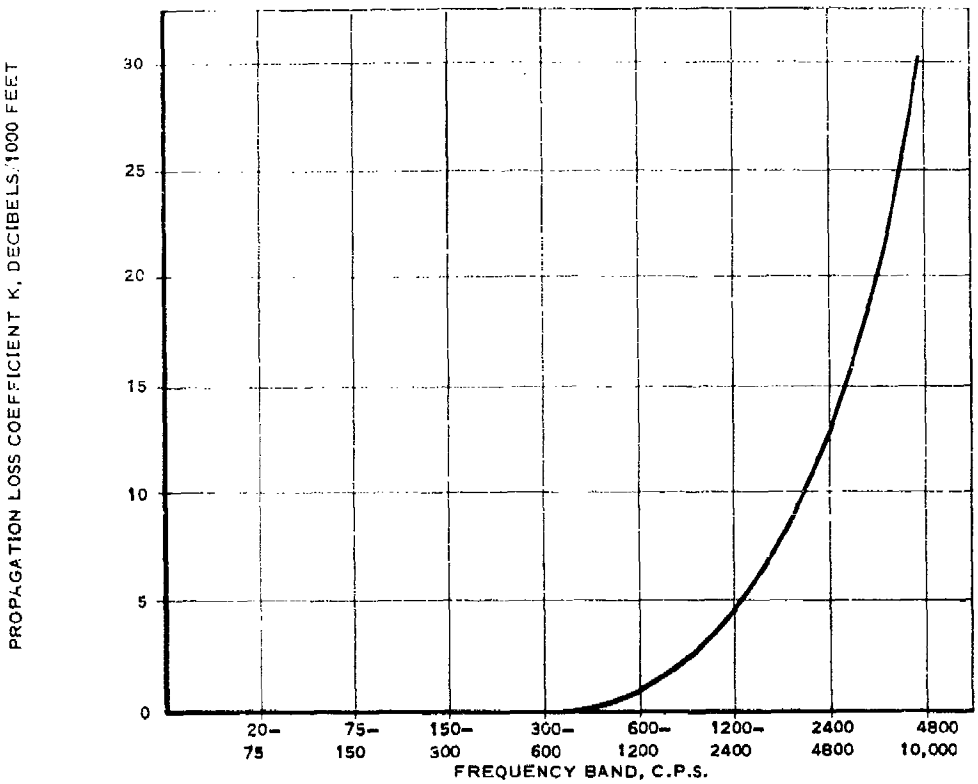


FIGURE 34. EFFECT OF ATMOSPHERIC ATTENUATION OF NOISE (REFERENCE 19).

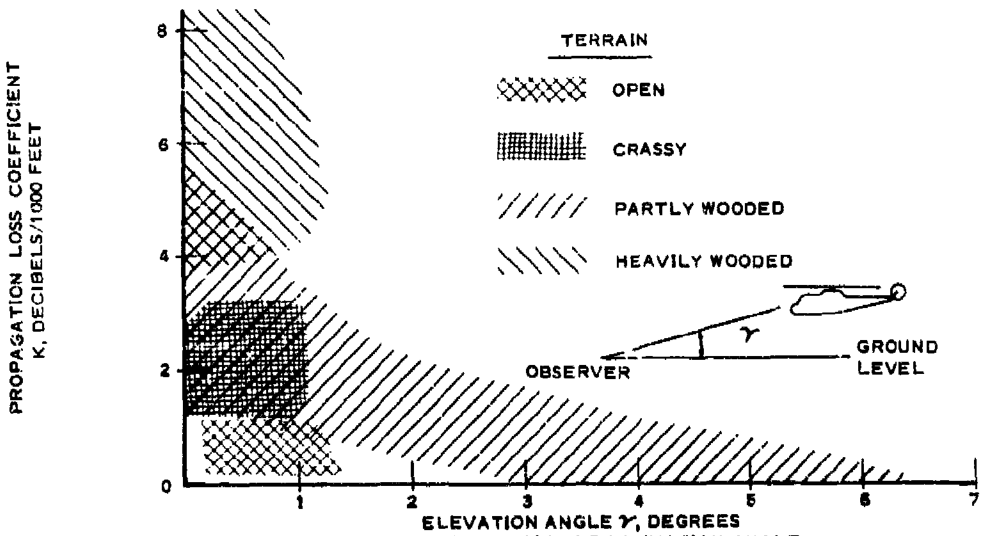


FIGURE 35. EFFECT OF TERRAIN AND ELEVATION ANGLE ON NOISE PROPAGATION (REFERENCE 19).

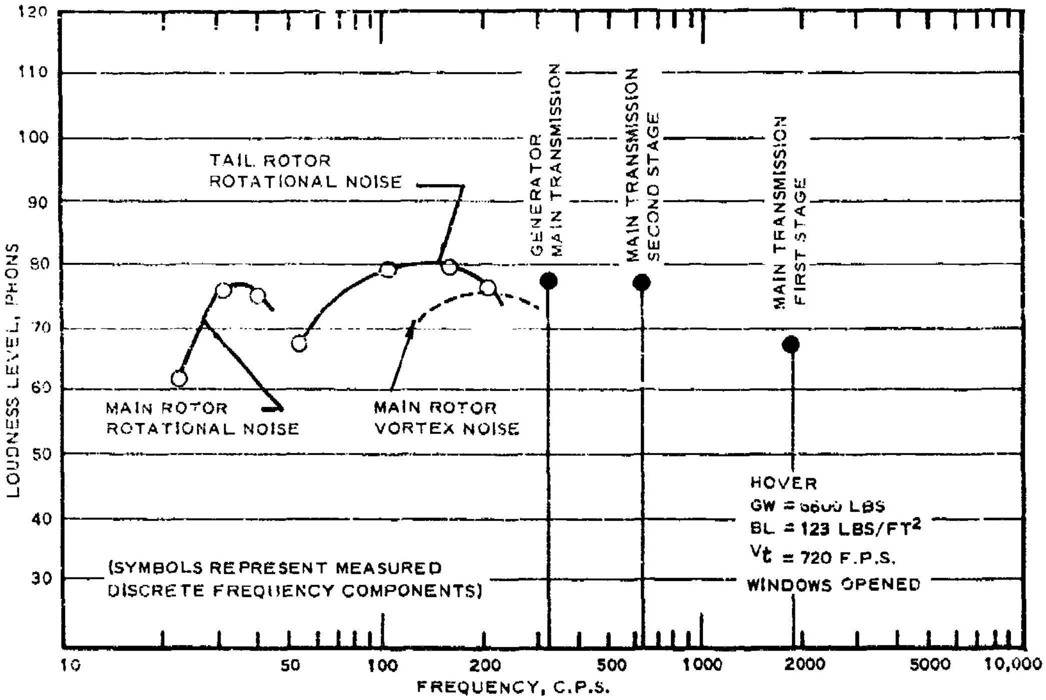


FIGURE 36. LOUDNESS LEVEL OF MAJOR HU-1A INTERNAL NOISE SOURCES.

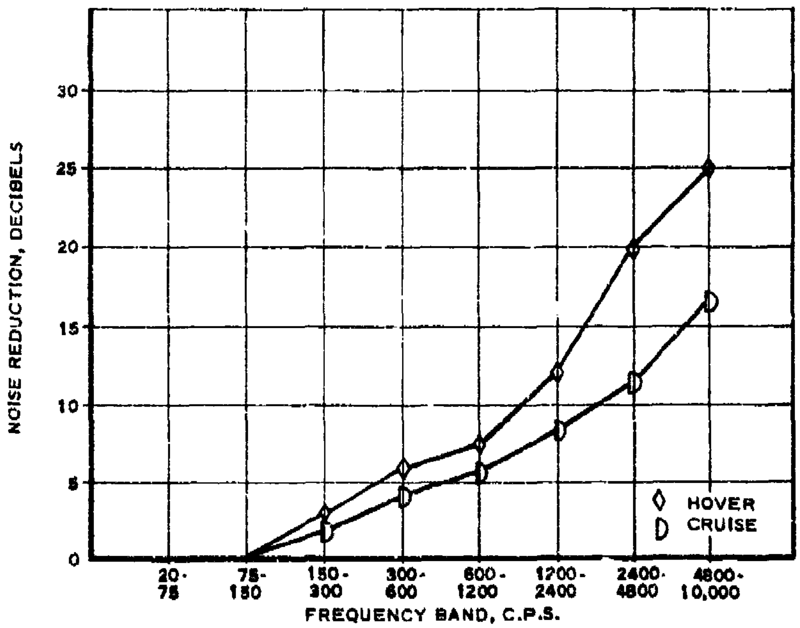


FIGURE 37. POSSIBLE HU-1A INTERNAL NOISE REDUCTION WITH ACOUSTICAL TREATMENT (REFERENCE 20).

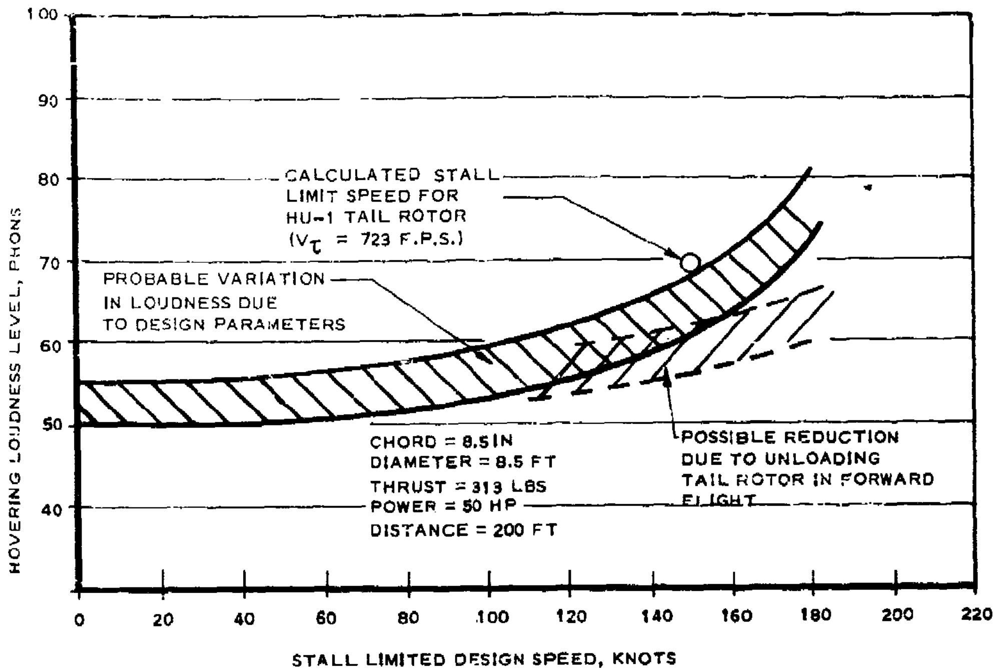


FIGURE 38. CALCULATED TAIL ROTOR LOUDNESS LEVEL AS A FUNCTION OF STALL LIMITED DESIGN SPEED.

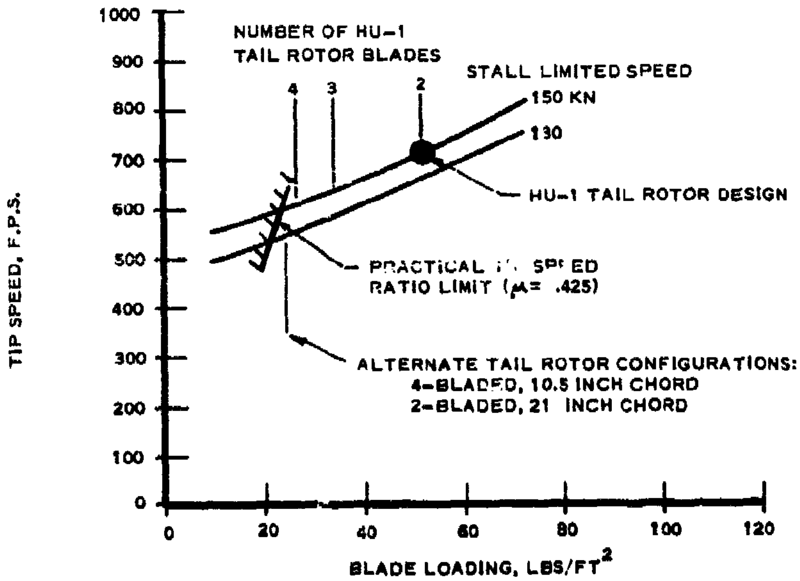


FIGURE 39. TAIL ROTOR TIP SPEED VERSUS BLADE LOADING FOR CONSTANT STALL LIMITED SPEED.

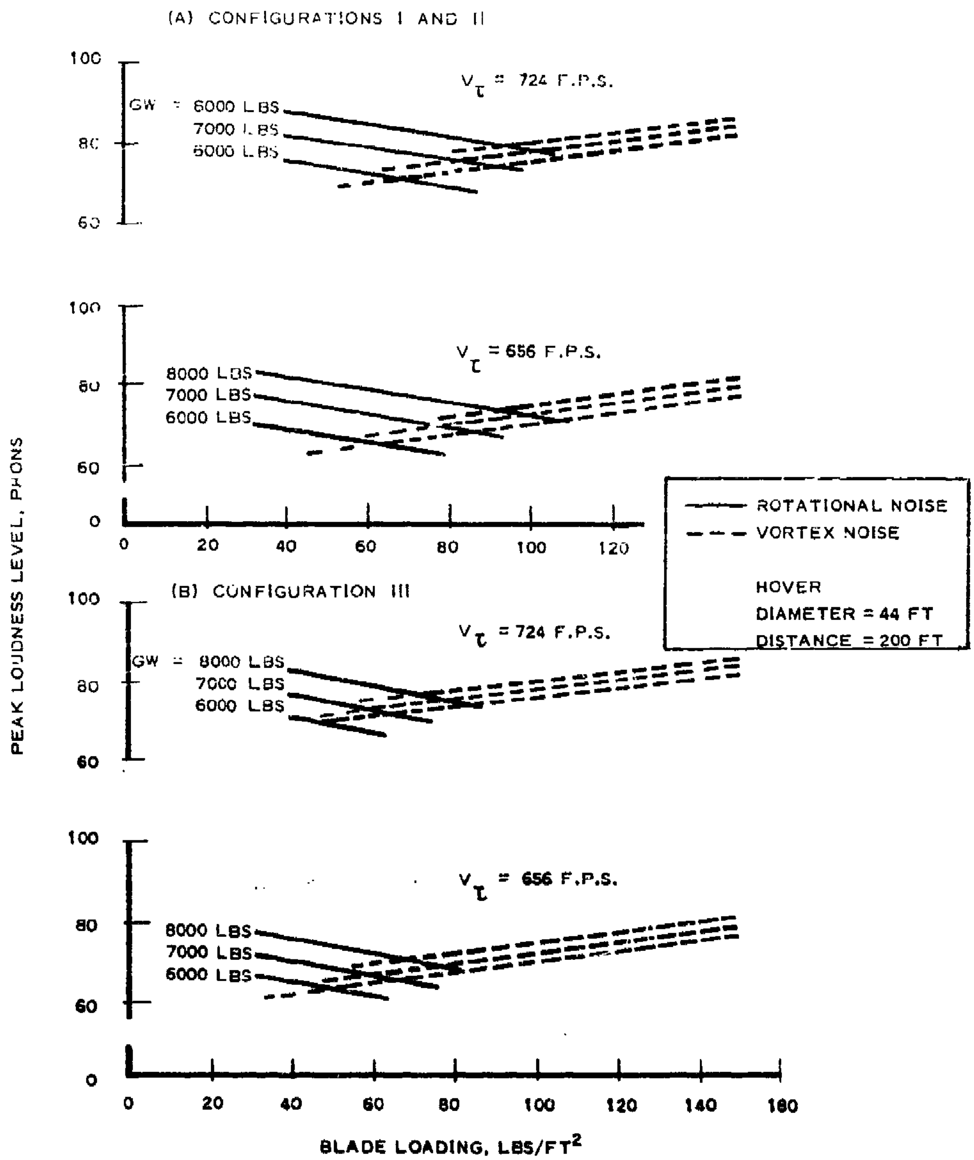


FIGURE 40. SUMMARY OF MEASURED MAIN ROTOR LOUDNESS LEVELS.

APPENDIX

COMBINED ACOUSTICAL AND DYNAMIC AIR LOAD DATA

Presented in the following pages are portions of the basic main rotor air load data which were recorded during the subject program. Data for two flight conditions are given: a 60-knot level flight and a 60- to zero-knot deceleration maneuver. In Figures 41 through 43 the differential blade pressures measured at 40, 75, 85, 90 and 95 per cent radius, respectively, are presented. Noted on each plot are chord locations of the various transducers and corresponding trace identification numbers. The over-all internal sound pressure level measured by portable acoustical equipment is given by trace number eight (8) in Figure 42.

The trace sensitivities based on a reference calibration pressure are given in Table 11. The trace numbers, the location of the transducers, the trace zeros (reference line of zero ΔP), and the calibration constants for each radial position are noted. The calibration constants are given in differential pressure (p.s.i.) per inch of trace deflection.

Within the scope of the subject program, the location of the helicopter with respect to the ground plane microphones and the azimuth positions of the rotor could not be established during the fly-over tests. Therefore, acoustical data of ground plane microphones taken during these tests are not included.

TABLE 11
DIFFERENTIAL PRESSURE TRACE SENSITIVITIES

RADIAL POSITION	TRACE NUMBER	CHORD LOCATION Per Cent	TRACE ZERO Inches	CALIBRATION CONSTANT Psi/Inch
40% R	6	4	.05	1.35
	7	17	.75	.71
	8	34	1.72	.43
	9	63	2.14	.16
	10	85	3.72	.10
75% R	1	2	-.01	1.80
	2	9	.23	1.60
	3	17	.41	1.16
	4	23	.85	1.09
	6	63	2.07	.23
	7	90	2.87	.13
85% R	1	2	-.05	3.40
	2	4	.12	3.60
	3	9	.64	2.82
	4	13	.76	1.95
	5	17	1.13	4.07
	6	23	3.57	3.08
	7	34	3.13	1.57
	9	63	2.38	.37
	10	77	3.71	.37
	11	90	4.71	.36
90% R	1	2	-.03	3.88
	2	9	.54	3.65
	3	17	1.70	4.07
	4	23	.38	2.05
	5	34	1.81	1.53
	6	63	3.30	.62
	7	90	4.78	.30
95% R	1	2	.08	3.87
	2	9	.77	5.38
	3	17	1.97	4.22
	4	23	1.22	2.11
	6	63	2.45	.48
	7	90	4.55	.17

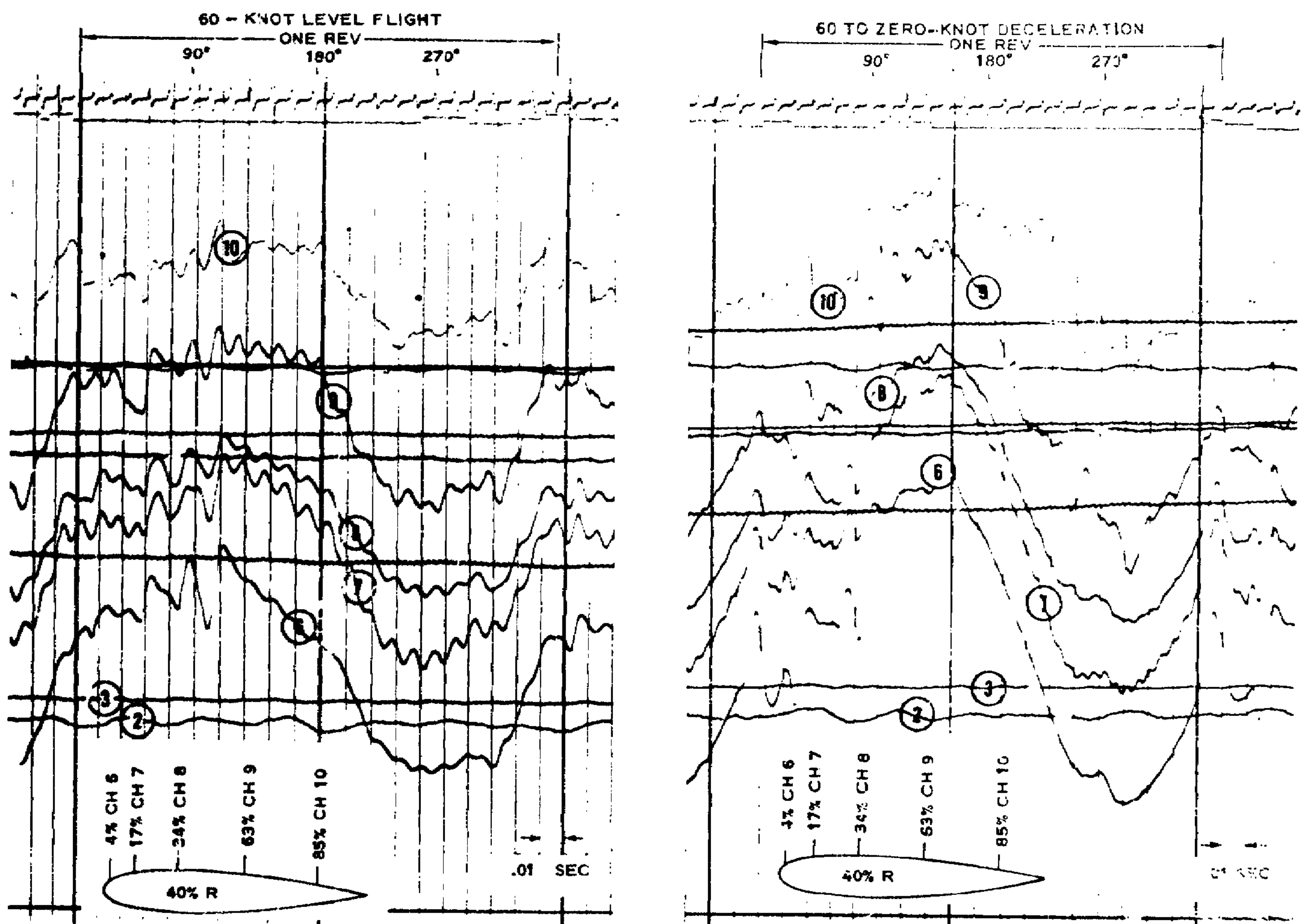


FIGURE 41. BLADE DIFFERENTIAL PRESSURE–40 PER CENT RADIUS.

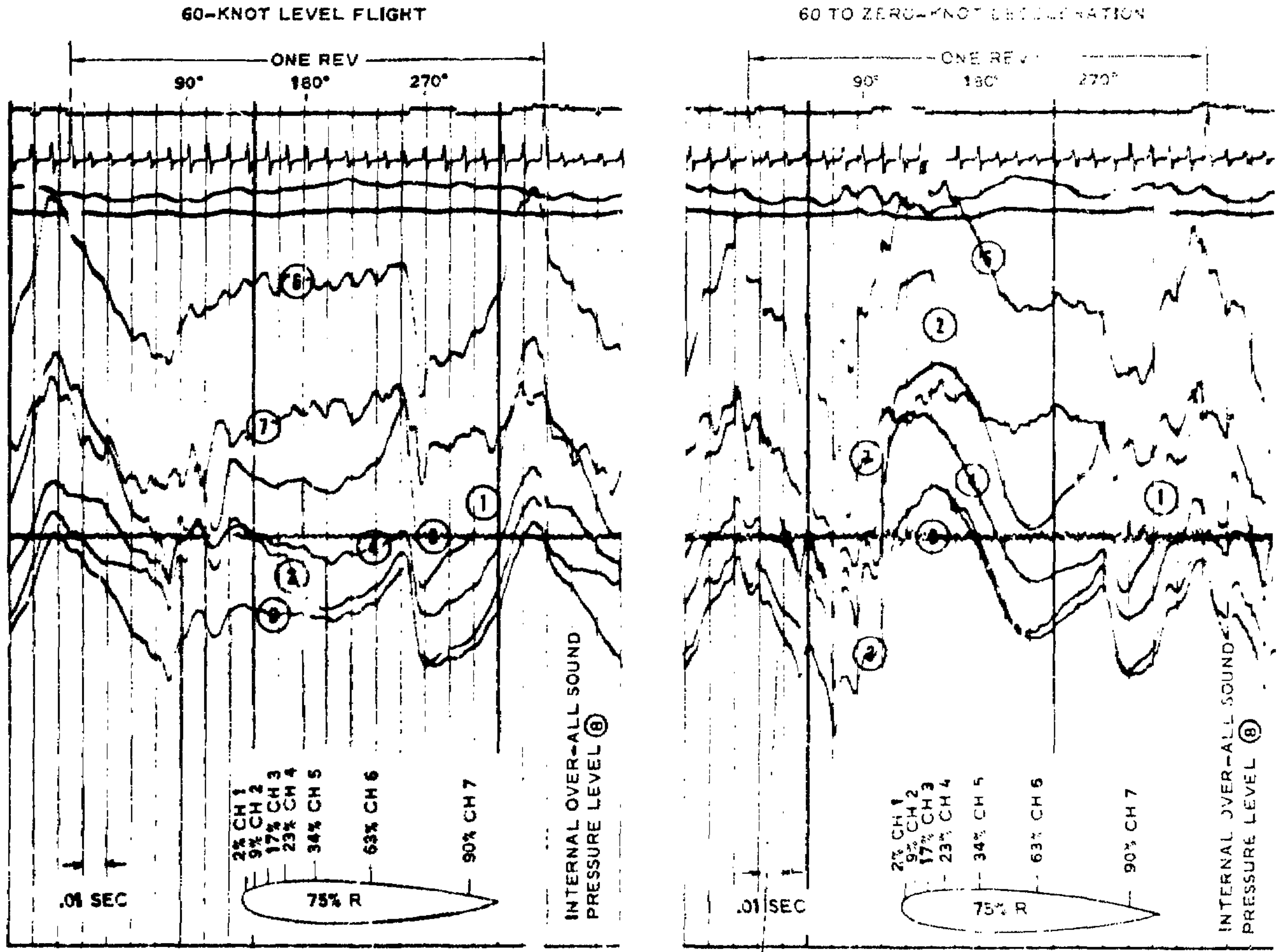


FIGURE 42. BLADE DIFFERENTIAL PRESSURE - 75 PER CENT RADIUS

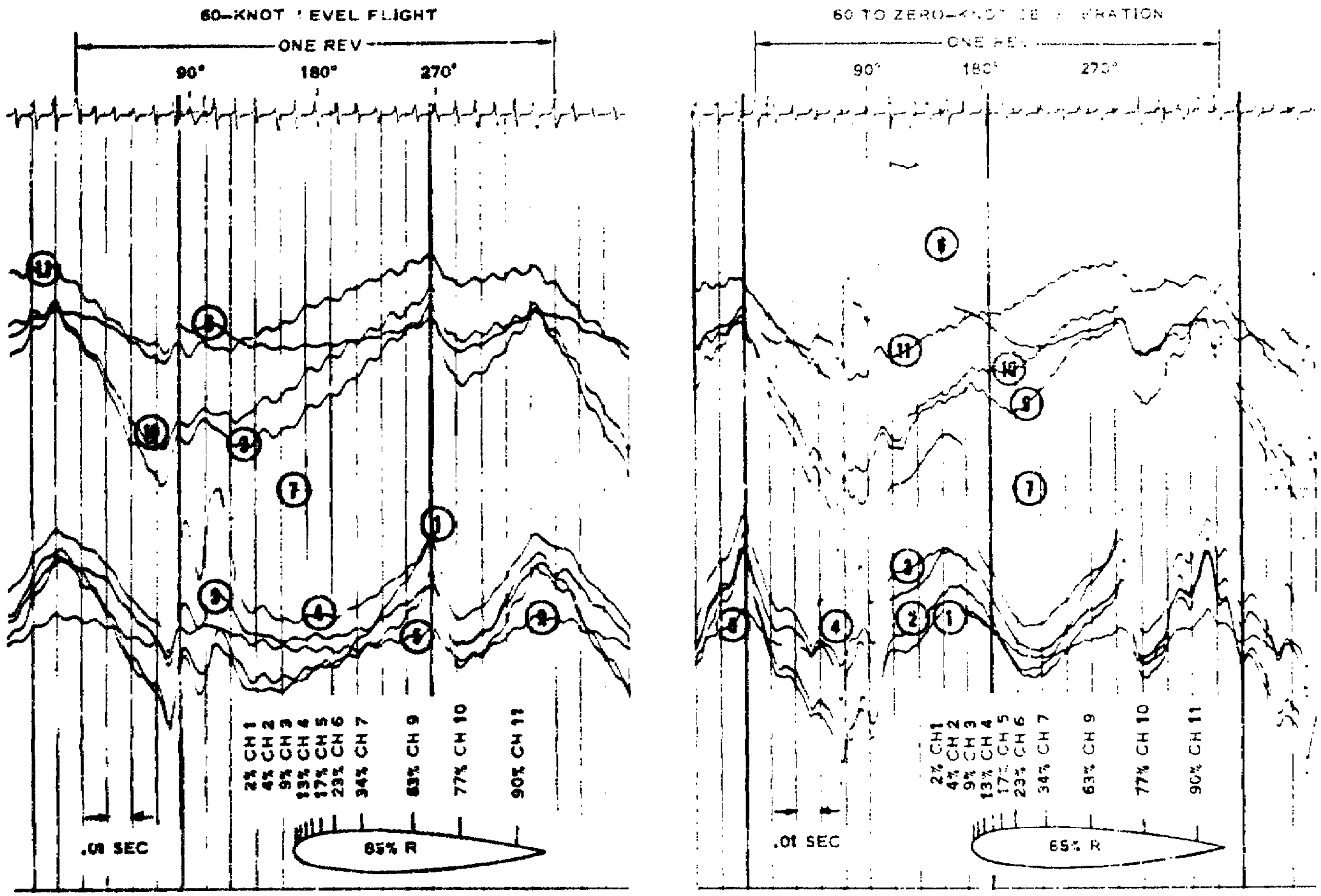


FIGURE 43. BLADE DIFFERENTIAL PRESSURE–85 PER CENT RADIUS.

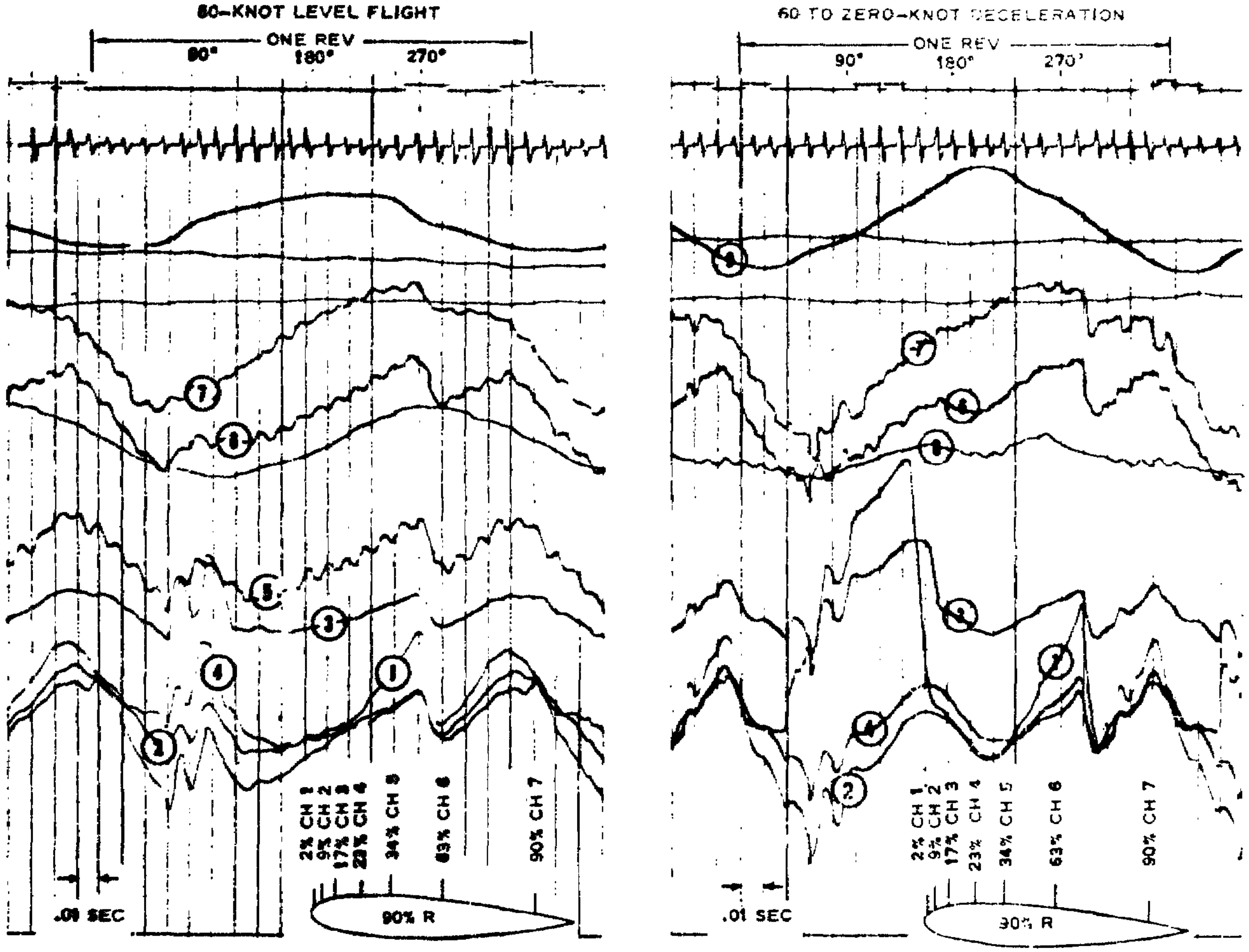


FIGURE 44. BLADE DIFFERENTIAL PRESSURE–90 PERCENT RADIUS.

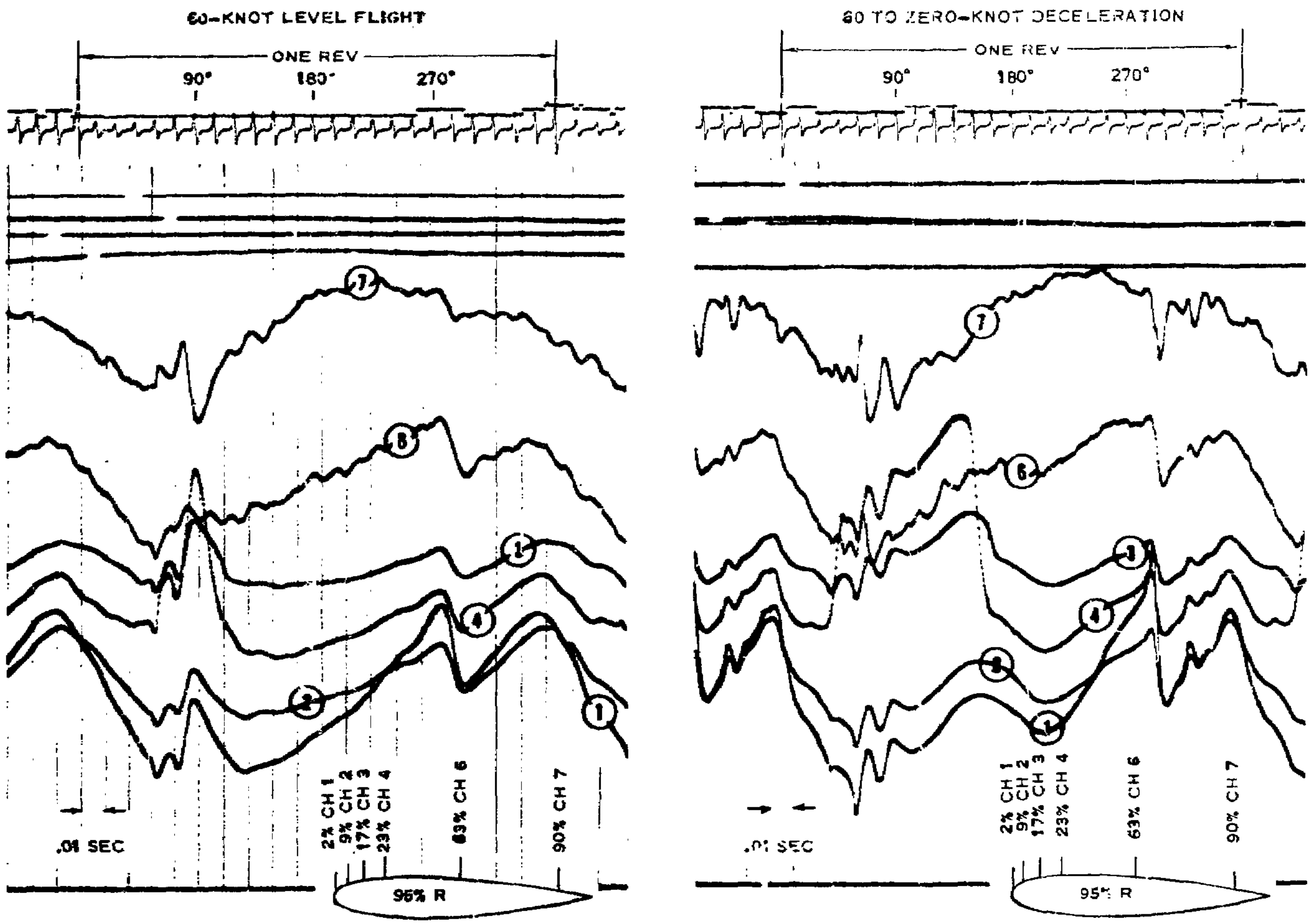


FIGURE 45. BLADE DIFFERENTIAL PRESSURE–95 PER CENT RADIUS.

DISTRIBUTION

USCONARC	3
First US Army	3
Second US Army	2
Third US Army	2
Fourth US Army	1
Sixth US Army	1
USAIC	2
USACGSC	1
USAWC	1
USAATBD	1
USAARMBD	1
USAAVNBD	1
USAPRDC	1
DCSLOG	2
Rsch Anal Corp	1
ARO, Durham	2
OCRD, DA	1
USATMC Nav Coord Ofc	1
NATC	2
CRD, Earth Scn Div	1
USAAVNS, CDO	1

DCSOPS	1
OrdBd	1
USAQMCDA	1
QMFSA	1
CECDA	1
CofT	6
USATCDA	1
USATB	1
USATMC	20
USATC&FE	4
USATSCH	3
USATRECOM	58
USA Tri-Ser Proj Off	1
USATTCA	1
TCLO, USAABELCTBD	1
USASRDL LO, USCONARC	2
USATTCP	1
OUSARMA	1
USATRECOM LO, USARDG(EUR)	1
USAEWES	2
TCLO, USAAVNS	1
USATDS	5

USARPAC	1
EUSA	1
USARYIS/IX CORPS	2
USATAJ	6
USARHAW	3
ALFSEE	2
USACOMZEUR	3
USACARIB	4
AFSC (SCS-3)	1
APGC (PGAPI)	1
Air Univ Lib	1
AFSC (Aero Sys Div)	2
ASD (ASRMPT)	1
CNO	1
CNR	3
BUWEPS, DN	5
ACRD (OW), DN	1
BUY & D, DN	1
USNPGSCH	1
CMC	1
MCLFDC	1
MCEC	1

MCLO, USATSCH	1
USCG	1
Lewis Rsch Cen, NASA	1
Sci & Tech Info Fac	1
USGPO	1
ASTIA	10
HUMRRO	2
US Patent Ofc, Scn Lib	1
ASD, FCL	1
MOCOM	3
USSTRICOM	1
Bell Helicopter Co.	10
NAFEC	3
Langley Rsch Cen, NASA	2
Geo C. Marshall Sp Flt Cen, NASA	1
MSC, NASA	1
NASA, Wash., D. C.	6
Ames Rsch Cen, NASA	2

Bell Helicopter Company,
Fort Worth, Texas.
A STUDY OF THE ORIGIN AND MEANS OF REDUCING HELICOPTER NOISE - C. R. Cox and R. R. Lynn
Report No. 299-099-180, November 1962, 102 pp - illus - tables (Contract DA 44-177-TC-729) Task 9R 38-01-022-01, TCREC Report 62-73.
Unclassified Report

An experimental and analytical investigation of helicopter noise and means for its reduction are presented.
(over)

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2. VTOL/STOL Aircraft, Helicopter Noise Reduction
3. Contract DA 44-177-TC-729

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